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QUIET CLEAN SHORT-HAUL EXPERIMENTAL ENGINE (QCSEE)

Preliminary Over-the-Wing Flight Propulsion System Analysis Report

by

D.F. Howard, et al

GENERAL ELECTRIC COMPANY

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FOREWORD

The Quiet Clean Short-Haul Experimental Engine (QCSEE) Prog· ˥ is currently being conducted by the General Electric Company, Aircraft Engine Group in accordance with NASA Contract NAS3-18021, under the direction of Mr. C.C. Ciepluch, NASA Project Manager. The Program includes the design, manufacture, and test of an under-the-wing (UTW) and an over-the-wing (OTW) experimental engine. Both engines are intended to develop the technology needed for externally blown flaps, short takeoff and landing, commercial, short-haul aircraft.

To ensure the selection of appropriate flight system parameters and characteristics, and to provide design guidance, subcontracted study support was obtained from aircraft manufacturers and operators. General Electric selected The Boeing Company to provide support for the OTW system, Douglas Aircraft Company to provide support for the UTW system, and American Airlines to evaluate both installations and provide an appropriate operational scenario. Specific subcontracted effort consisted of guidance in selection of the engine cycles, installation design, propulsive-lift interactions, control interfaces, acoustics, performance, and economic analyses.

Although earlier studies had indicated a need to operate from a 609.6 m (2000 ft) runway, it was concluded by all contributors that the flight studies of a commercial short-haul transport should be conducted based on a 914.4 m (3000 ft) runway, typical of existing close-in airports.

The experimental system retained the 609.6 m (2000 ft) runway requirement to assure the technology margin for the aircraft ready to enter airline service in the mid 1980's. In either case, the propulsion system would be designed to meet a noise requirement of 95 EPNdb at 152.4 m (500 ft) sideline during approach and takeoff. Since final system requirements will not be defined for some time yet, the experimental engine objectives [including 609.6 m (2000 ft) runway], being the more stringent, were not changed. Thus the technology margin that is developed in the QCSEE Program will be adequate for any forseeable system requirement.

This report covers the subcontracted analyses of the OTW aircraft system. Comparable analyses of the UTW aircraft systems are described in Reference 1. The propulsion systems used in these studies were projected "flight" systems based on the technology being developed in the experimental program. Propulsion system weight, performance, and installation features have been projected on a rational basis from the experimental propulsion system design.

Design studies were performed to develop concepts for integrating the flight design version of the QCSEE propulsion system with the OTW, externally blown flap (EBF), powered lift aircraft designed for short-haul service. The airplane concept was based on technology consistent with providing a reliable vehicle that is both durable and economical to operate and that could be

ready to enter airline serivce in the mid 1980's. The preliminary design studies produced definitions of the airplane configuration, characteristics, performance, and operating economics.

The airplane accessories requirements were identified and the concept of an accessories pack installed remote from the engine accessories gearbox was developed. The space envelope, weight, and significant features and characteristics of the various components were identified as were the estimates of the shaft power demands this system placed on the engine power takeoff (PTO) drive. Studies were made to develop the concepts for the cabin environment control and lifting surface ice protection systems needed for the short-haul airplane. The air bleed demands imposed on the engines were estimated and concepts for matching the bleed capacity to the demands were developed.

The conclusions that could be drawn from the results of the work have been documented.

Significant contributions to this report were made by the Boeing Commercial Aircraft Company in the areas of aerodyanmics, advanced design, avionics, environmental control, power plant, structures, and weights. American Airlines provided the short-haul aircraft requirements and conducted various installed reviews of the propulsion system.

This report deals exclusively with the QCSEE OTW Flight Propulsion System design and analyses based on the aircraft use of a 914.4 m (3000 ft) runway. With the longer runway, the aircraft will attain a higher takeoff velocity permitting reduced aircraft flap angles and reduced engine thrust. On approach, the longer runway permits higher aircraft approach velocity, also reducing flap angle and thrust requirements. This results in a reduction in the amount of noise suppression panel treatment required to meet the acoustic program objective.

1.0 SUMMARY

The Quiet Clean Short-Haul Experimental Engine (QCSEE) Program includes the preliminary design and installation of high bypass, geared turbofan engines with nacelles forming the propulsion systems for short-haul, passenger aircraft. These flight systems include the technology required for an externally blown flap type aircraft with over-the-wing (OTW) propulsion system installations for introduction into passenger service in the mid 1980's.

Based on the flight designs, the program provides for the design, fabrication, and testing of an OTW experimental engine containing the required technology items for low noise, fuel economy, "D"-shaped exhaust nozzle and digital engine control. The design of the experimental OTW engine is described in Reference 2.

This report summarizes the preliminary design of the QCSEE OTW Flight Propulsion System installation and nacelle component and systems design features of a short-haul, powered lift aircraft. A substantial portion of this report was produced at The Boeing Commerical Aircraft Company and covers their efforts in support of the QCSEE program.

The major purpose of the QCSEE Program is to develop and demonstrate the technology required for propulsion systems for quiet, clean, and economically viable commercial short-haul aircraft. This comprehensive program includes the following objectives:

- To develop the propulsion system technology which will permit a short-haul aircraft to achieve the system noise goal of 95 EPNdB along a 152 m (500 ft) sideline when the engines are scaled to a total installed thrust of 400,300 N (90,000 lb). The design shall also minimize the ground area (footprint) exposed to objectionable noise levels.

- To demonstrate a propulsion system which will meet advanced pollution goals under all operating conditions.

- To develop the technology for very-high-bypass ratio engines with quiet low-pressure-ratio geared variable-pitch fans.

- To develop the technology required to meet propulsion system performance, control, weight, and operational characteristics.

- To develop the material, design, and fabrication technology for quiet propulsion systems which will yield engine designs which have an uninstalled thrust-to-weight ratio greater than 6 to 1 and installed thrust-to-weight ratios greater than 3.5 to 1.

- To develop the technology which will yield engine thrust response characteristics required for powered lift operations.

- To provide the technology which will permit the design of quiet, efficient, lightweight thrust reversing systems for powered lift aircraft.

- To provide the technology to permit the design of integrated engine and nacelle installations which will be tolerant to aerodynamic distortions expected with operating flight conditions (such as high crosswinds, large angles of attack, and side slip) and still provide good cruise performance.

- To provide the digital electronic engine control technology required to improve engine and fan pitch control, thrust response, operational monitoring, and relief of some of the pilot's workload especially during the powered lift flight operations in the terminal area.

2.0 INTRODUCTION

This report presents the preliminary results of activities conducted under the Supporting System Design and Economics Studies task of the Quiet Clean Short-Haul Experimental Engine (QCSEE) program. The primary objectives of contract tasks 1.1.2, 10.1, and 10.2 are to provide design guidance to the experimental engine based on flight installation system studies and to update the evaluation of a conceptual flight propulsion system design based on QCSEE test results. This report covers the QCSEE Preliminary OTW Flight Propulsion System Analysis during the period of the detail design of the OTW experimental engine.

The aircraft system, economic, and installation studies were conducted by the Boeing Commercial Airplane Company as a subcontractor to General Electric. The representative short-haul airliner was evolved from past Boeing studies including "Quiet Propulsive Lift Research Aircraft Design Study" (QSRA) conducted under NASA contract NAS2-7951. These studies show that high bypass, low pressure ratio turbofan engines have the potential of providing an economical propulsion system for achieving the very quiet aircraft noise level of 95 EPNdB on a 152.4 m (500 ft) sideline.

The OTW engines, with low exhaust velocities associated with the high bypass ratio and the shielding effect of the wing, result in very low community noise exposure. Numerous advanced technology items are included in the QCSEE program such as electronic engine controls, airflow control with a modulating nozzle, integrated engine nacelle structure, and near-sonic inlet for noise reduction. These items require a higher degree of engine-to-airframe integration than provided by current design approaches.

The specific study aircraft was designed to the requirements specified by American Airlines in the document "Operational Scenario and General Requirements for Multi-Engine STOL Passenger Transport Airplane for Introduction in 1980-1982" dated February 13, 1974; revised April 10, 1974 (see Appendix A). The resultant airplane carries 200 passengers to a design range of 925 km (500 N Mi) or 169 passengers to an extended range of 1387 km (750 N Mi) from a design field length of 914.4 m (3000 ft). The aircraft was sized to four QCSEE flight engines with an uninstalled takeoff thrust of 93408 N (21,000 lb) [SLS, 346 K (90° F)] per engine. Economic studies showed a direct operating cost of 2.29 cents per available seat statute mile at the design range based on 1975 pricing.

A concept for integrating a flight version of QCSEE with this airplane has been evaluated. Completed activities include those that address structural interfaces such as engine/wing mounting and parting surfaces, accessory locations, and basic nacelle internal and external aerodynamic lines.

3.0 AIRCRAFT - BASELINE

3.1 DESIGN REQUIREMENTS

The principal design requirements for the baseline OTW aircraft are:

Payload - 200 passengers

Range - 925 km (500 N Mi)

Field Length - 914.4 m (3000 ft) for S.L., 346 K (90° F) conditions

Noise - No greater than 95 EPNdB on 152.4 m (500 ft) sideline

An extended range capability of 1387 km (750 N Mi) with a full passenger payload is desired when operating from runways longer than 914.4 m (3000 ft). These design requirements are based on specifications by American Airlines (Appendix A).

A four-engine OTW airplane designed to these requirements requires an engine thrust of 68,944 N (15,500 lb). This engine is considerable smaller than the QCSEE design and would require development of a new core, substantially increasing the price of a new engine. The F101 core is expected to have a long production run, which results in a more economical engine. Therefore, aircraft system and economic studies were conducted on the basis of the QCSEE flight engine, designated GE19F4E2, which uses a modified F101 core. Since this engine produces 93,408 N (21,000 lb) uninstalled thrust, the baseline aircraft is to be sized to 914.4 m (3000 ft) field length and 925 km (500 N Mi) mission range, resulting in a passenger payload capability in excess of the design requirements. The payload can be off-loaded for additional fuel to meet the 1387 km (750 N Mi) extended range mission.

3.2 SIZING METHODS

The baseline aircraft was sized from parametric data developed for the QSRA study (Reference 3). The aerodynamic and propulsive lift effects in Reference 3 were obtained from model tests adjusted for parametric variations in aircraft configurations. Inputs to the airplane sizing program were modified to include the QCSEE propulsion weights, thrust, fuel consumption, aerodynamics, and variations in passenger payload. Rather than reanalyze the aircraft for the many design variables of sweep, aspect ratio, thickness ratio, etc., these parameters were selected from the optimized values reported in Reference 3. The design characteristics of the QCSEE aircraft were then established by the sizing process described below.

The sizing method outlined in Figure 1 involves the process of calculating combinations of wing loading (W/S) and thrust-to-weight ratio (T/W) which satisfy a 914.4 m (3000 ft) takeoff field length. Parametric aircraft were sized to the 925 km (500 N Mi) range for each of these combinations. The computerized airplane sizing program "thumbprint" then determines the point design characteristics of a minimum gross weight aircraft using the mission rules shown in Figure 2. The characteristics of this design were determined in terms of gross weight, payload, wing area, block fuel, DOC, etc. This process was repeated by varying cruise altitude and Mach number to establish the effect of these variables on DOC and block fuel. The resulting baseline airplane characteristics were then determined on the basis of minimum DOC, fuel utilization, and operational factors.

The landing field lengths of the designs were also computed to determine compatibility with the 914.4 m (3000 ft) takeoff field length. For the range of variables investigated, landing field length was found to be less than takeoff requirement.

The design criteria used in the sizing process is described in the following paragraphs.

Takeoff and landing performance was based on FAR Part 25 and Part XX. The takeoff field length is defined as the greater of:

- 1.15 times the all engine takeoff to a 10.7 m (35 ft) height
- Distance to 10.7 m (35 ft) height with critical engine failure speed at V_1
- Distance to accelerate to V_1 and decelerate to a stop.

The performance margins are based on those proposed in Reference 4. This results in the following criteria for takeoff, approach, and landing:

3.2.1 Takeoff

Minimum control speed on the ground (V_{MC_G}) = 28.3 m/sec (55 knots) with critical engine inoperative (CEI).

Rotation speed = 1.05 V_{MC_G}

Liftoff speed (V_{LO}), greater of:

- 1.08 minimum unstick speed (V_{MU})
- 1.05 mininum control speed in air (V_{MCA}) with CEI

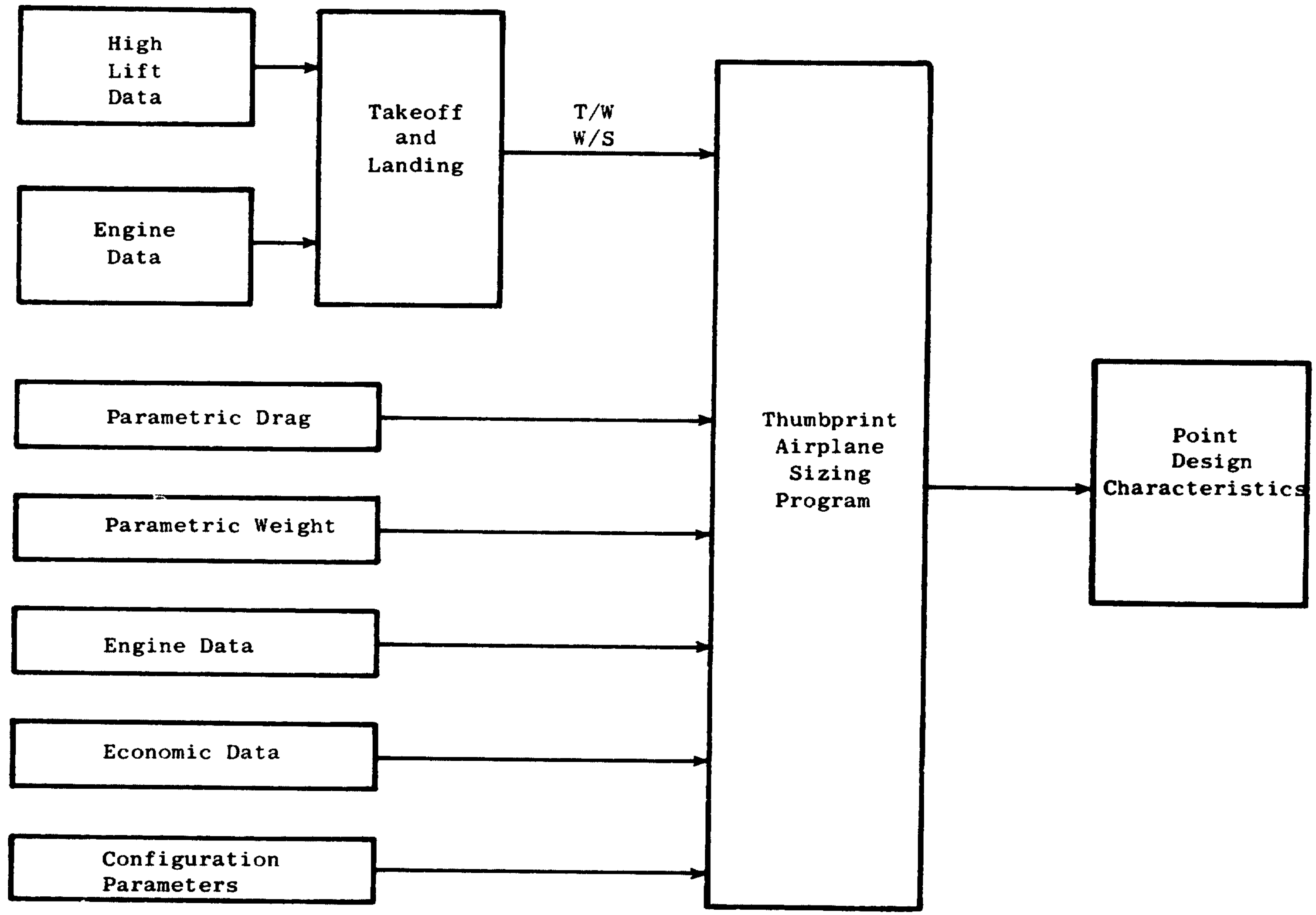

Figure 1. Sizing Process.

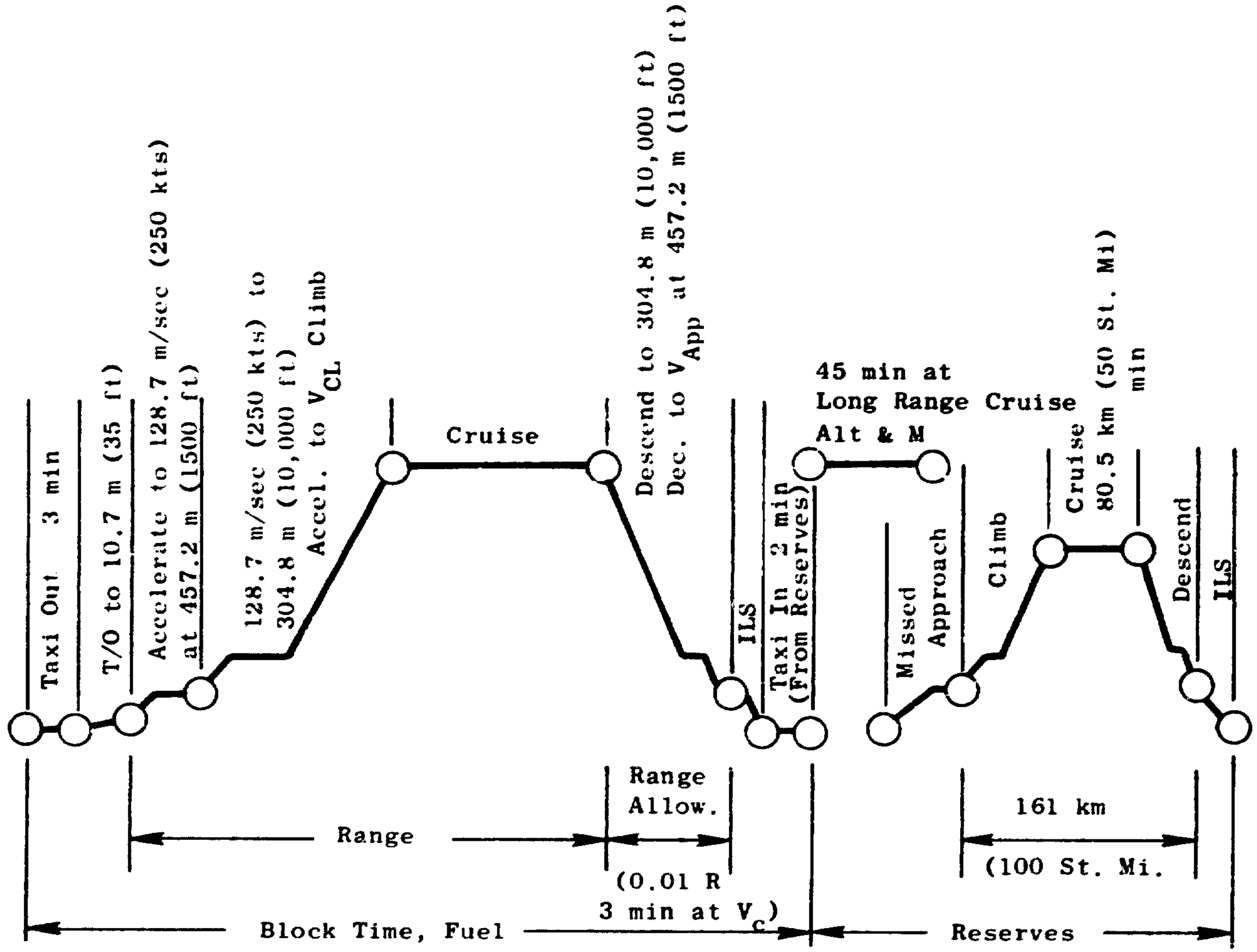

Figure 2. STOL Mission Rules.

Climbout speed (V_2), greater of:

- 1.10 V_{MCA}
- 1.15 minimum speed (V_{MIN}) for steady-state flight at maximum thrust with CEI or AEO
- 0.25 g maneuver margin with CEI or all engines operative (AEO)

Climbout gradient (γV_2) = 0.03 with CEI

Rolling coefficient of friction = 0.015

Braking coefficient of friction (μ_B) = 0.4

Transition time = 1 sec (throttles, brakes, and spoilers)

3.2.2 Approach and Landing

Approach speed, greater of:

- Speed such that approach can be safely continued without change of configuration with an engine failure.
- 1.15 V_{MIN} AEO or CEI at approach thrust

    (V_{MIN} is minimum speed for steady-state flight at approach thrust level.)

    $V_{MIN} > V_{MC} = 28.3$ m/sec (55 knots)
- 1.10 V_{MC_A} CEI at maximum thrust

Go-around climb gradient, equal to or greater than:

- 0.032 at V_{APP} (AEO at approach flaps) or 1.2 m/sec (240 ft/min), whichever is greater.
- 0.027 at V_{APP} (CEI with go-around flaps) or 1.2 m/sec (240 ft/min), whichever is greater.

Maneuver capability (Δg), equal to or greater than:

- 0.25, AEO or CEI at V_{APP} and approach thrust

Angle attach margin from stall ($\Delta\alpha$), greater of:

- 10°, CEI at V_{APP} and approach thrust
- 15°, AEO at V_{APP} and approach thrust or CEI at V_{APP} and maximum thrust

Braking coefficient of friction (μ_B) = 0.4

The landing field length is defined as 1.67 times the landing distance, which is the distance to clear a 10.7 m (35 ft) height and come to a complete stop. The landing distance is composed of three segments:

1. Air Distance - Distance from 10.7 m (35 ft) height (threshold) to touchdown with flare initiated at 9.1 m (30 ft) and approach speed.
2. Transition Distance - Distance covered in 1 second delay after touchdown to the full application of thrust reduction, brakes, and spoilers.
3. Braking Distance - Distance from full brake application (μ_B = 0.4) to a complete stop.

3.3 AIRCRAFT PERFORMANCE

The QCSEE baseline OTW aircraft is capable of transporting a maximum payload of 17,872 kg (39,400 lb) to the design range of 925 km (500 N Mi) from a 914.4 m (3000 ft) sea level field on a 346 K (90° F) day. The aircraft is designed for 197 passengers on the basis of 90.7 kg (200 lb) per passenger with luggage. The aircraft can carry 169 passengers [15,332 kg (33,800 lb payload)] to the extended range of 1387 km (750 N Mi) from the same field length as shown by the payload range capability on Figure 3. The increased fuel weight required for the longer range is compensated by off-loading payload to maintain a maximum takeoff gross weight of 90,040 kg (198,500 lb). If all payload is exchanged for fuel, a 4352 km (2,350 N Mi) ferry range can be achieved. Thus, the baseline aircraft exceeds payload requirement at the design stage length stated in section 2.1 and can perform the extended range mission without requirement for longer runways.

Although the aircraft is designed for a complement of 197 passengers, it is also possible to transport cargo if the passenger payload is reduced by the corresponding cargo weight. As an example, 4264 kg (9400 lb) of cargo and 150 passengers can be transported to the 925 km (500 N Mi) range.

Takeoff performance of the aircraft shown on Figure 4 is predicated on operating all engines at maximum thrust with the nozzle side doors open. The USB flaps are retracted with the first and second segments of the outboard flaps set at 36 and 58 degrees, respectively. The aircraft clears a 10.7 m (35 ft) height at an airspeed of 58.4 m/sec (113.5 knots) in a distance of 796 m (2610 ft) from brake release and climbs along a 0.16 gradient at constant airspeed. During climbout, the engine centerline is at an attitude

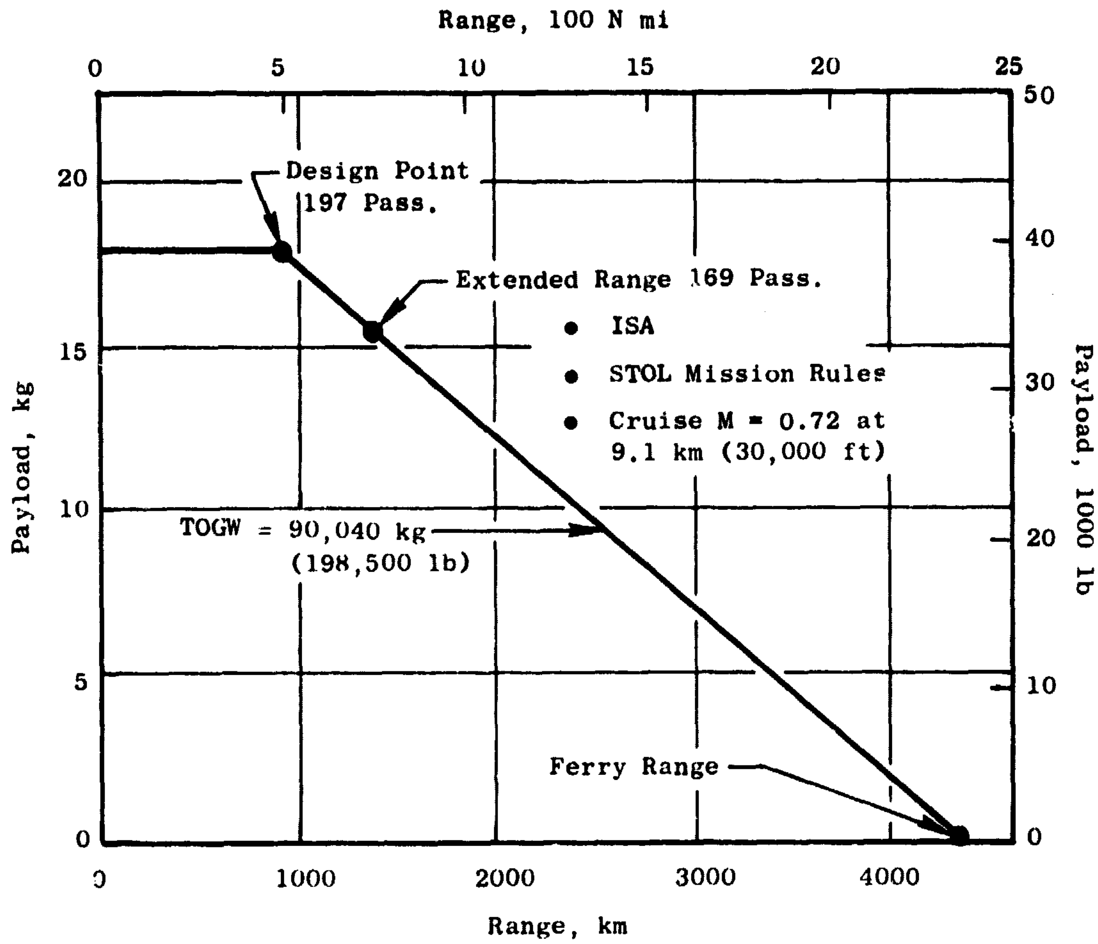

Figure 3. Mission Performance.

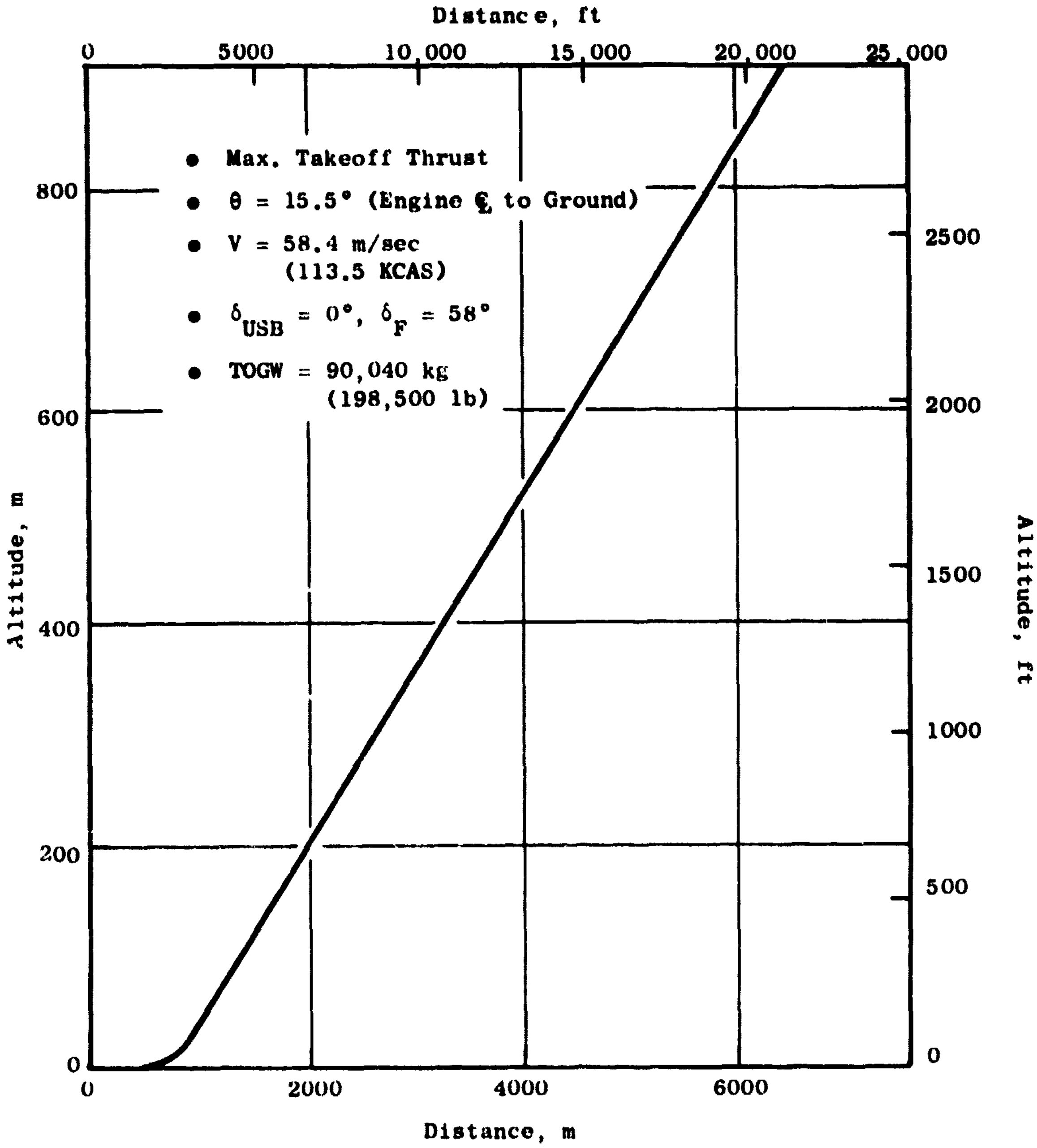

Figure 4. Takeoff Climb Profile.

of 15.5 degrees with respect to the ground. The velocity history is shown in Figure 5. The aircraft continues to climb at maximum thrust until a 9.1 km (30,000 ft) cruise altitude is reached; at which point the engines are throttled to 88% thrust to establish 0.72 Mach number with the nozzle side doors fully closed.

The landing performance of the aircraft is based on an approach path of 6 degrees and a landing gross weight of 83,553 kg (184,200 lb) at completion of the 925 km (500 N Mi) mission as shown on Figure 6. In the approach configuration, the first and second segment of the USB flaps are set at 23 and 40 degrees chordal angle, respectively, with the outboard flaps at 36 and 58 degrees, respectively. A nominal approach speed of 47.8 m/sec (93 knots) is maintained at 50% maximum engine thrust with the nozzle side doors in the open position. During approach, the engine centerline will be at an attitude of -2 degrees with respect to the ground. After crossing the 10.7 m (35 ft) threshold height, a ground distance of 121.9 m (400 ft) is covered until touchdown, followed by a ground roll of 274.3 m (900 ft) for a total landing distance of 396.2 m (1300 ft).

3.4 AIRCRAFT DESCRIPTION AND CHARACTERISTICS

The OTW aircraft general arrangement shown in Figure 7 utilizes the upper surface blowing (USB) technique to provide propulsive lift. High flow turning angles and efficiencies can be achieved by spreading the exhaust flow over the upper wing surface, causing the engine efflux to thin and flow over the trailing edge flaps.

To implement the USB principle, the engines are mounted forward of the wing leading edge with the upper portion of the exhaust nozzle pointing toward the wing surface. During takeoff and approach, the nozzle side doors are open to achieve lateral flow spreading which is then turned by double-slotted flaps. During normal engine operation, the USB flap slots are sealed to provide a smooth turning surface and to reduce the radiated noise below the aircraft. The portion of the double-slotted flaps outboard of the USB flaps are unsealed and are operated independently. The trailing edge flaps in conjunction with full-span, variable-camber leading edge flaps provide a very high propulsive lift configuration.

During approach with a critical engine out, the trailing edge USB flap slots behind the inoperative engine are opened and the flaps further extended. The USB flaps behind the operative engine are partially retracted. This technique, in addition to inboard location of the engines, considerably reduces the engine-out rolling moment which allows the control system to trim the airplane with less drag.

An advanced technology airfoil used on an 8.5 aspect ratio wing permits the use of a low quarter chord sweep of 6.5 degrees at a cruise Mach number of 0.72. The primary wing structure is a single box with integral fuel tanks which results in a 29,711 kg (65,500 lb) maximum fuel capacity to provide long range ferry flight capability.

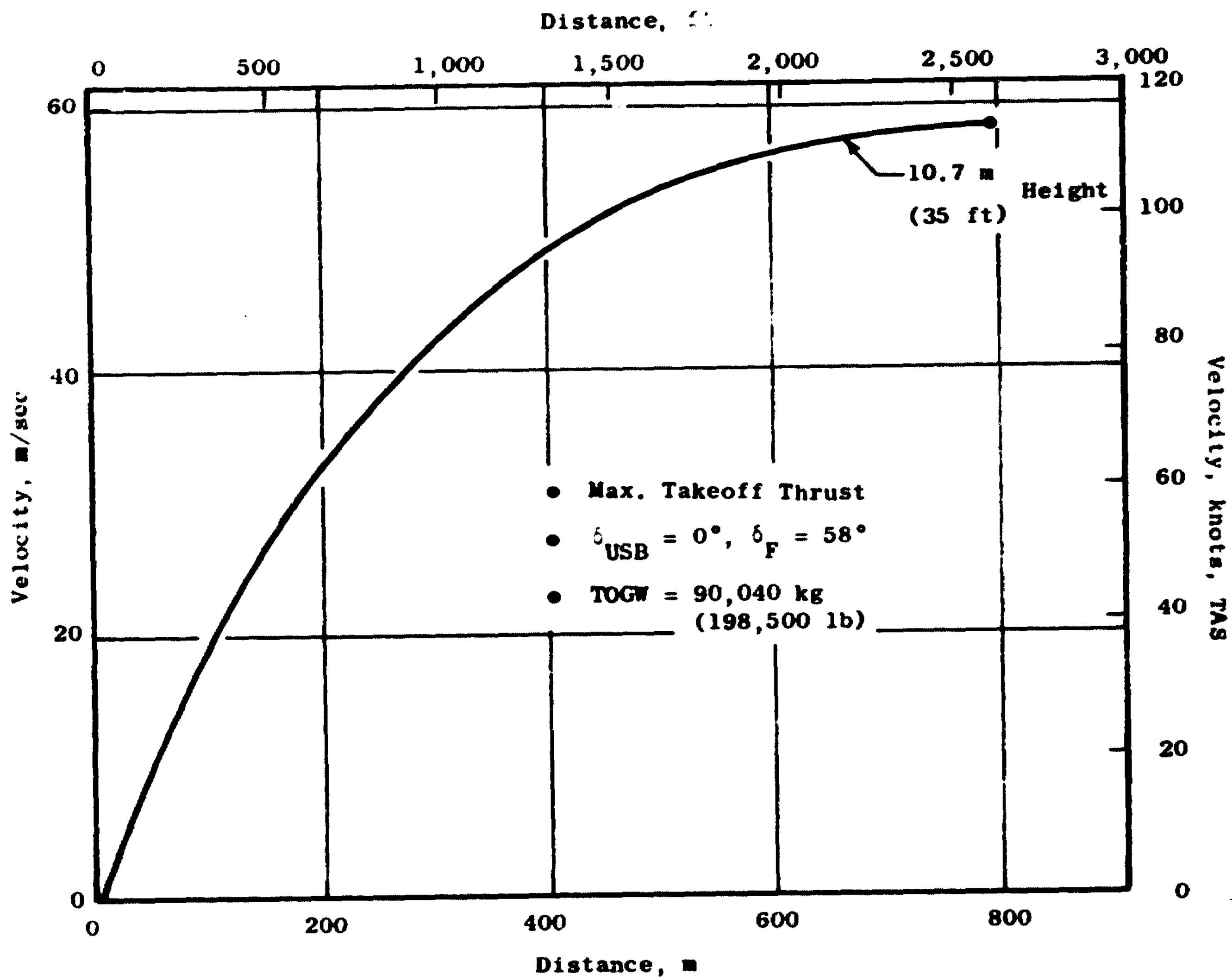

Figure 5. Takeoff Velocity.

- V_{APP} = 47.8 m/sec (93 KCAS)
- δ_{USB} = 23°/40°, δ_F = 58°
- 50% Max. Takeoff Thrust
- W = 83,740 kg (184,620 lb)
- θ = -2° (Engine ℄ to Ground)

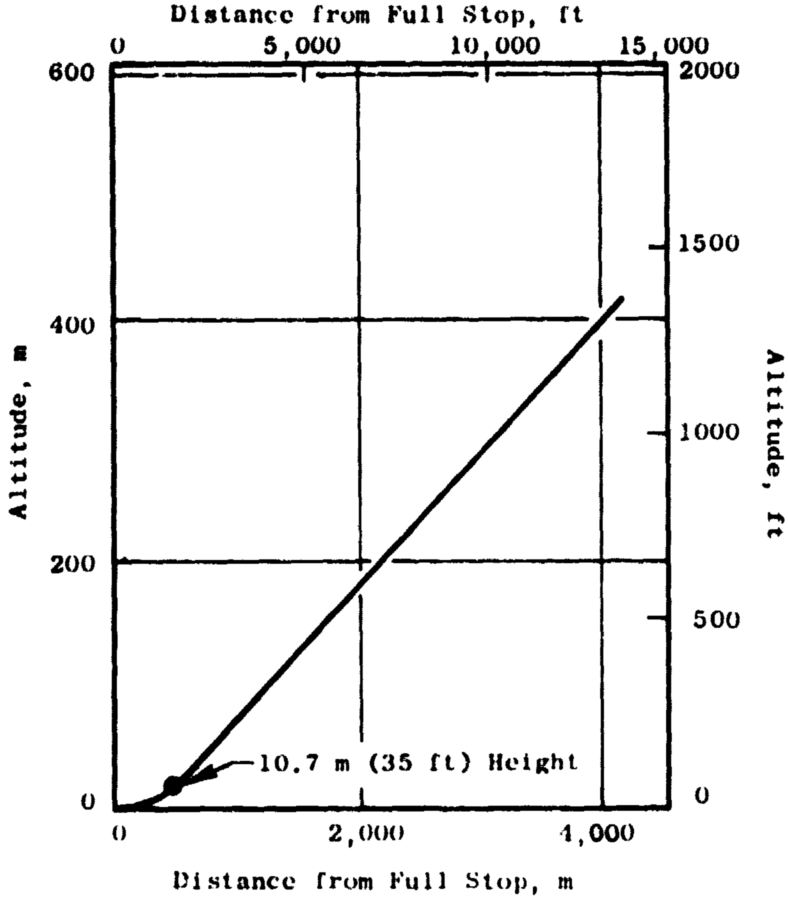

Figure 6. Approach Profile.

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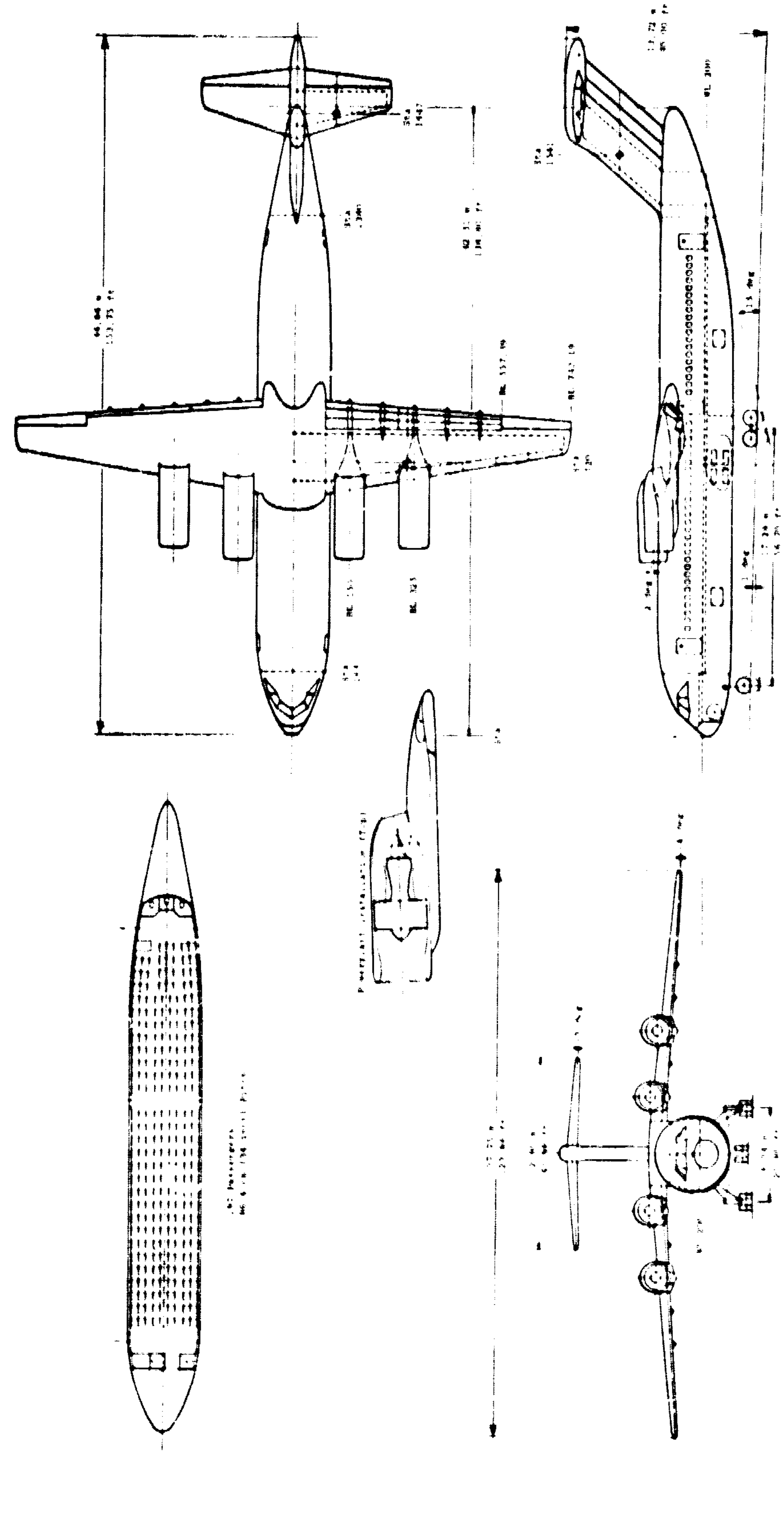


Figure 7. Baseline OTW Aircraft General Arrangements.

The wing is set at an incidence of 2 degrees with respect to the fuselage, and the engine centerline is placed at an angle of -2 degrees with respect to the fuselage for a total of 4 degrees between wing and engine centerline. This aligns the nacelle with the local airstream during cruise, providing appropriate angles for an efficient cruise configuration as well as a level cabin floor.

The aircraft employs spoilers and ailerons for lateral control. Directional and longitudinal control are accomplished by means of double-hinged rudders and elevators, respectively. The horizontal stabilizer is mounted high on the vertical tail to minimize tail size. The spoilers are also used as a direct-lift control and the USB flaps are modulated for speed control.

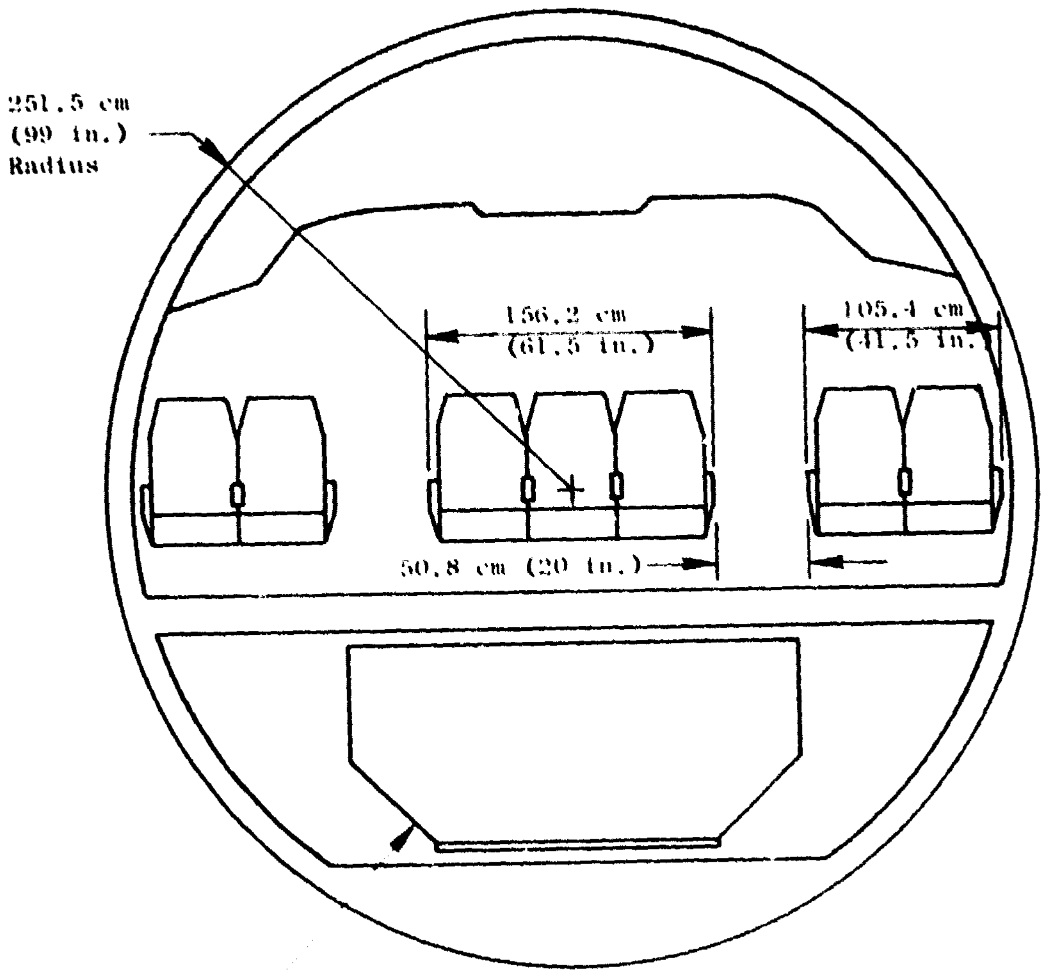
Due to the large payload capability of the four-engine OTW airplane, a wide-body fuselage with seven abreast seating is necessary to limit body length for adequate rotation at takeoff. The interior arrangement shown in Figure 7 is designed for short-haul, single-class service. A buffet-coffee bar is located at each end of the cabin, since hot food service is considered impractical for short-haul operations. Provisions are made for lavoratories, coat racks, and attendant stations. A double-aisle seating arrangement, in conjunction with 106.7 cm (42 in.) wide doors located at each end of the passenger compartment, is used to minimize passenger unloading and loading time. A 86.4 cm (34 in.) pitch seating with 50.8 cm (20 in.) wide aisles, shown in Figure 8, provides a seating comfort level similar to the 747 wide-body aircraft. The 502.9 cm (198 in.) diameter body provides 90.6 m^3 (3200 ft^3) of under-floor cargo volume foreward and aft of the main gear stowage compartment. This volume can be used for containerized and bulk cargo.

A blocker door installed in the upper portion of the USB nozzles is used for reverse thrust. The engine exhaust is directed up and forward which reduces hot gas reingestion and engine damage due to foreign objects. This also places a down load on the landing gear, increasing the aircraft braking effectiveness during ground roll. The thrust reverser can be used to back away from airport ramps. This, in conjunction with airstairs at each end of the passenger compartment, materially reduces the ground support equipment required by the aircraft.

A summary of the basic OTW aircraft characteristics is shown in Table I.

3.5 INSTALLED PROPULSION PERFORMANCE

The installed performance of the QCSEE GE19/F4E2 flight engine was obtained by a combination of General Electric and Boeing estimates. GE provided the primary and secondary mass flows, temperatures, and pressures at the exit of the primary nozzle (mixing plane) after accounting for flow path losses, power extraction and airbleed for various flight conditions as outlined in Figure 9. An ideal thrust was computed by Boeing using the GE data which was then multiplied by an adjusted C_V to obtain the installed engine thrust. The procedure is described in Figure 10.



LD-3 Base Container

Figure 8. Body Cross Section.

Table 1. OTW Aircraft Characteristics.

Weights	Gross Weight	Payload	Operating Empty Weight (OEW)
	90,040 kg (198,500 lb)	17,872 kg (39,400 lb)	60,420 kg (133,200 lb)
Surfaces	Wing	Horizontal	Vertical
Area m^2 (ft^2)	167.7 (1805)	40.9 (440.5)	37.8 (352.9)
Aspect Ratio	8.5	4	1
Taper Ratio	0.3	0.4	1
C/4 Sweep (deg)	6.5	11.5	40
Incidence (deg)	2	---	---
Dihedral (deg)	-4	-3	---
t/c (%)	18/16	16	16
Mac (C_{ref}), cm (in.)	486.9 (191.7)	339.3 (133.6)	572.5 (225.4)
Span, m (ft)	37.8 (123.86)	12.8 (41.98)	5.7 (18.75)
Tail Arm, cm (in.)	---	2355 (927)	2085 (821)
Tail Vol. Coef. ($\overline{V}$)	---	1.18	0.108
Body, cm (in.)	Length	Max Dia	---
	4232 (1666)	502.9 (198)	
Propulsion	Number	Type	Ins. Thrust
	4	GE19/F4E2	86,158 N (19,370 lb)
Landing Gear	Nose	Main	Loc. % C_{ref}
	2 40 x 13	8 42 x 7	50.1
Fuel Capacity	Wing*, kg (lb)	C.G. (% C_{ref})	---
	29,711 (65,500)	40	

*Excludes Body Center Section.

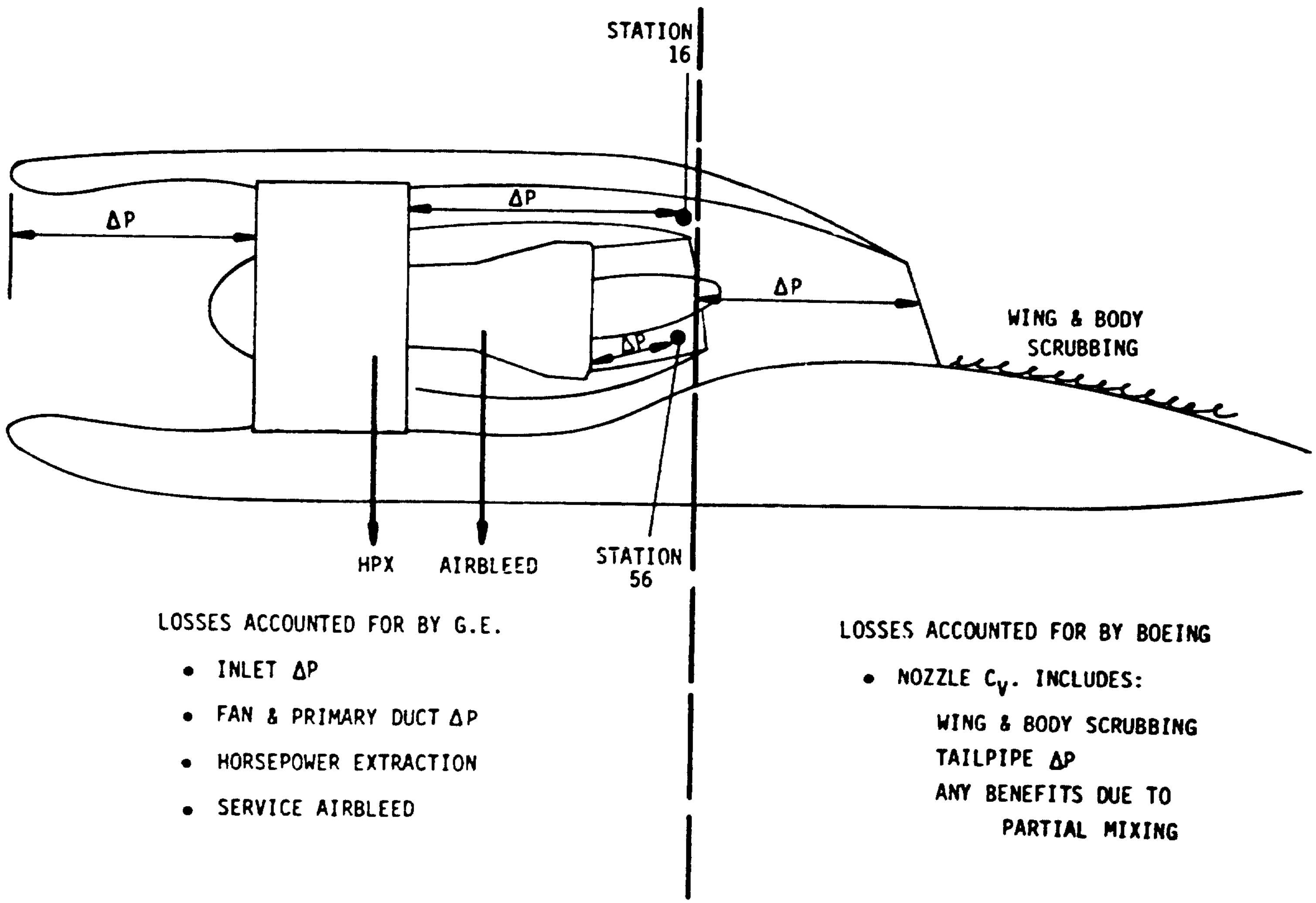


Figure 9. Accounting of Installation Losses.

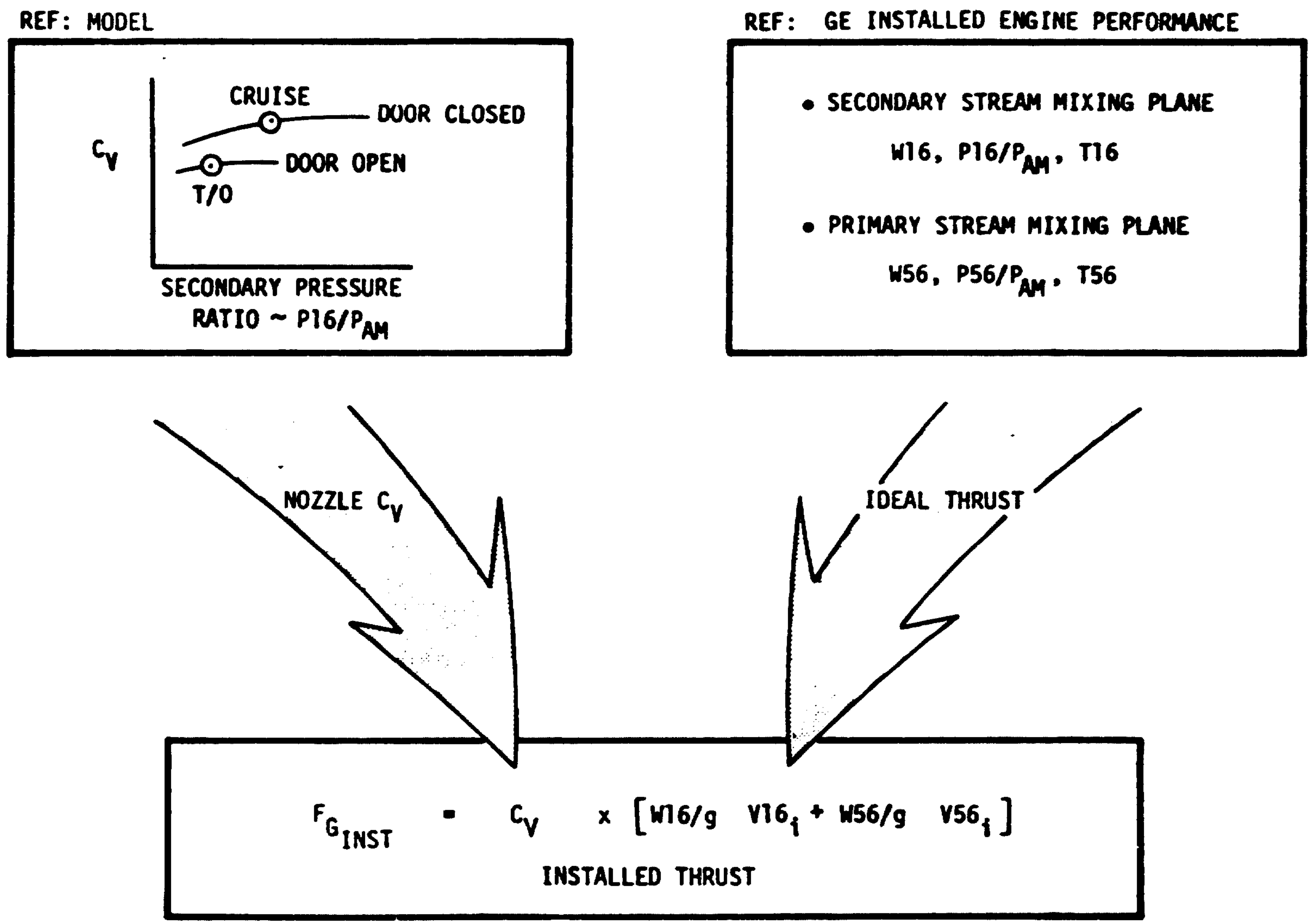


Figure 10. Installed Performance.

The nozzle C_V was estimated using technology developed from previous model tests of similar nozzles mounted on a wing body combination which employed upper surface blowing. These tests were conducted with hot primary and cold secondary airflows at various pressure ratios, measuring axial and normal force components on a balance. The ideal nozzle thrust was computed using measured mass flows, temperatures, and pressures at the primary and secondary stream mixing plane. Since the resultant thrust was known from the force measurements, the model velocity coefficient was obtained from the following relation.

$$F = C_V \left(\left[\frac{W}{g} V_{ideal} \right]_{pri} + \left[\frac{W}{g} V_{ideal} \right]_{sec.} \right)$$

This technique includes losses in the nozzle, external surface scrubbing, and benefits due to partial mixing of the streams.

Adjustments to the C_V obtained from the model tests were made for geometry differences. The basic differences are aspect ratio and nozzle overarea required by the QCSEE engine at takeoff. The aspect ratio of the QCSEE nozzle is 1.8 compared to 3.2 for the model. The lower aspect ratio accounts for an improvement in velocity coefficient of 0.003 for both takeoff and cruise.

The QCSEE nozzle requires a 20% overarea at takeoff for proper engine match, compared to 5% overarea used for model tests. The overarea for QCSEE is achieved by the use of a door on each side of the nozzle exit. During takeoff and approach the side doors are opened resulting in an increase in nozzle effective area and jet spreading. The 20% nozzle overarea results in a reduction in takeoff C_V of 0.01. The QCSEE nozzle C_V adjusted for the geometry differences described above are 0.951 and 0.985 for takeoff and cruise, respectively, as shown in Figure 11.

The resulting sea level takeoff thrust for a 346 K (90° F) day is shown in Figure 12 as a function of airspeed. The cruise thrust and sfc are shown in Figure 13 as a function of both altitude and Mach number on a standard day. The sfc shown in this figure was obtained by correcting the uninstalled values for the installed losses discussed above. The installed propulsion performance shown in these figures was used for the aircraft sizing and economic analysis studies.

3.6 WEIGHT

The weight data to determine the baseline aircraft were adapted from Reference 3. Since the payload for the reference study is predicated on 148 passengers, the parametric weight data were extended to include variations to accommodate larger passenger payloads and modified to incorporate the QCSEE flight propulsion system weight and its installation. The baseline aircraft weights for the QCSEE design conditions are shown in Table II.

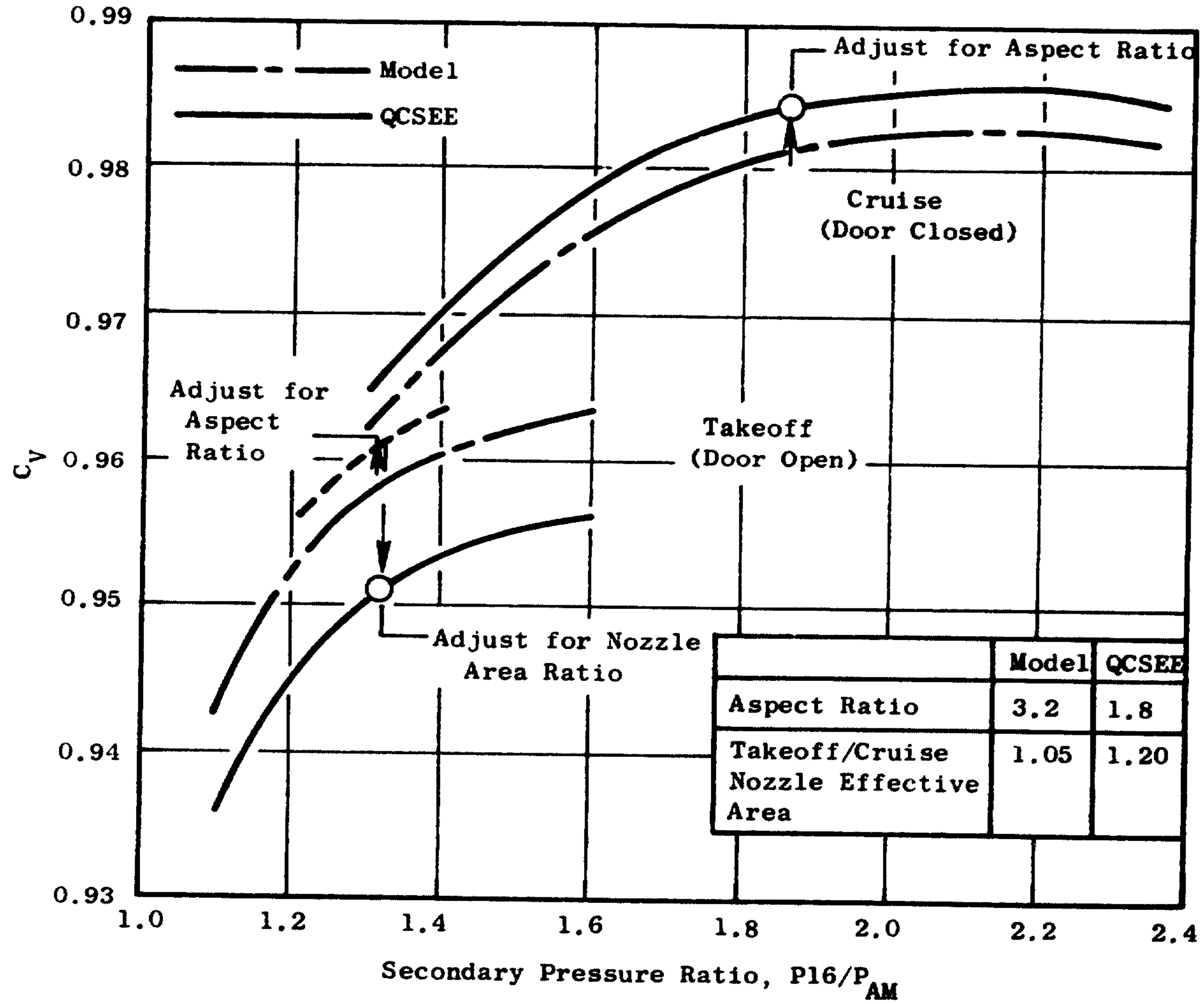


Figure 11. USB Nozzle Performance.

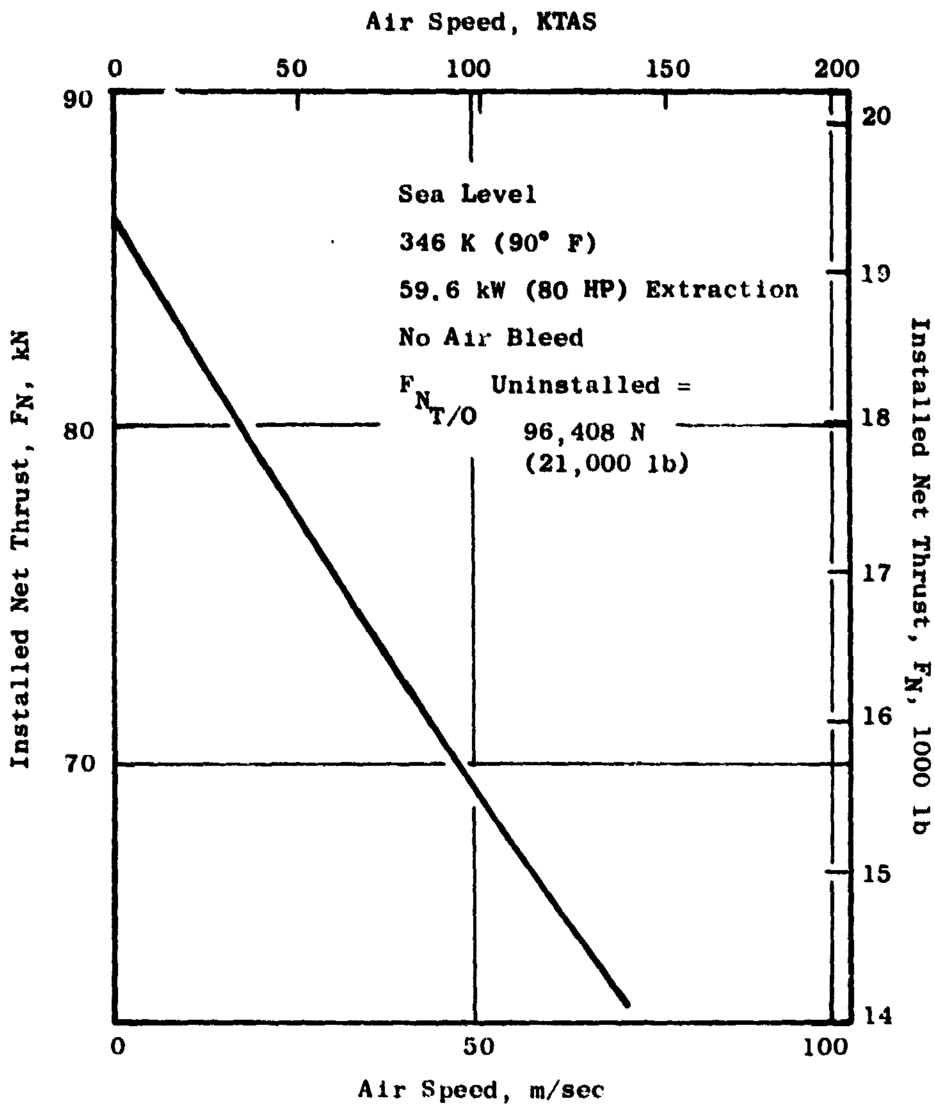


Figure 12. Installed Takeoff Thrust.

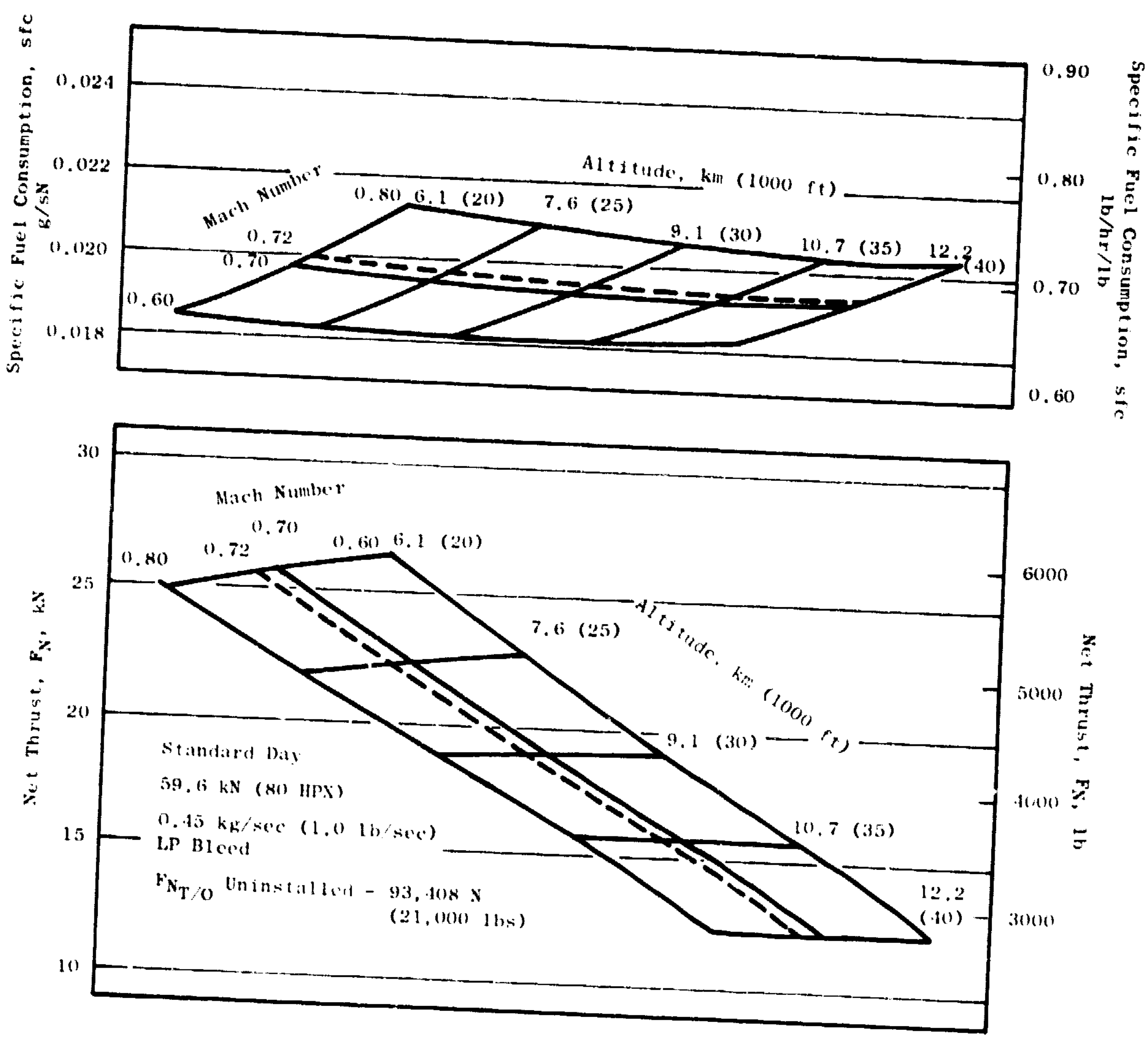


Figure 13. Installed Maximum Cruise Performance.

The weights for a long range ferry flight are also included. The propulsion weights are based on a GE estimate of 1943 kg (4284 lb) for an engine and nacelle, which includes the engine systems. A weight of 694 kg (1530 lb) per engine is estimated for propulsion installation. This is composed of two cantilevered struts and a shell mounted from the front spar, remote accessory drive, fittings, and wing upper surface heat shield.

Table II. Baseline Aircraft Weight Statement.

	kg	lb
Structure	35,980	79,320
Propulsion (Engine, Starting, Control, and Fuel System)	6,137	13,530
Equipment Systems	16,135	35,570
Manufacturer's Empty Weight	58,251	128,420
Standard and Operational Items	2,168	4,780
Operating Empty Weight (OEW)	60,420	133,200
500 N Mi Mission		
Payload (197 Passengers)	17,872	39,400
Fuel	11,748	25,900
Max. Takeoff Gross Weight	90,040	198,500
750 N Mi Mission		
Payload (169 Passengers)	15,332	33,800
Fuel	14,288	31,500
2350 N Mi Ferry Mission		
Payload	0	0
Fuel	29,620	65,300

4.0 PROPULSION SYSTEM/AIRPLANE INTEGRATION

4.1 REQUIREMENTS

The baseline short-haul airplane concept described in Section 3.0 is based on a propulsion system designed to meet the following requirements.

4.1.1 Thrust Requirements

The installed engine thrust requirements are presented in Table III. Installed thrust levels shall be consistent with the installation definition of Section 2.5 and shall include the power extractions specified in Section 3.1.2.

Takeoff and cruise thrust levels are those required to perform the airplane mission specified in Section 2.1. The ground idle thrust level is consistent with airplane braking requirements during ground operation.

Table III. Installed Thrust Requirements.

Maximum Takeoff Thrust Sea Level Static, 346 K (90° F)	86,158 N (19,370 lb)
Maximum Cruise Thrust 0.72 Mach, 9.1 km (30,000 ft), Std. day	18,904 N (4250 lb)
Ground Idle Thrust $\leq$4% of Takeoff Thrust	
Maximum Reverse Thrust $\geq$35% of Takeoff Thrust	

4.1.2 Power Extraction Requirements

Airbleed

A total of 1.8 kg/sec (4.0 lb/sec) of low pressure airbleed will be required to operate the airplane air conditioning and pressurization system for all flight conditions except during takeoff. During takeoff this airbleed requirement will be supplied by an inflight APU.

Horsepower Extraction

To provide power to operate hydraulic and electrical systems, 59.6 kW (80 shaft horsepower) per engine will be required.

4.1.3 Noise

95 EPNdB - 152.4 m (500 ft) sideline, during takeoff and approach.
100 PNdB - 152.4 m (500 ft) sideline, during maximum reverse thrust.

The above objectives are based on a four-engined 400,320 N (90,000 lb) SLS thrust aircraft.

4.1.4 Oil Consumption

0.9 kg/hr (2 lb/hr) maximum.

4.1.5 Dumping

No fluids shall be dumped under normal engine operation. Dumping may occur in case of abnormal operation such as seal failure.

4.1.6 Inlet Distortion

The system shall be stall free and otherwise satisfactory when operating at inlet angles of attack (α_i) up to and including 43° with approach M_0 of 0.18 [16.7 m/sec (120 knots)] and up to 50° at M_0 of 0.12 [41.2 m/sec (80 knots)] and with cross winds at 90° to inlet axis of up to 18 m/sec (35 knots). This enables the airplane to perform the flight demonstrations needed to certificate the type of airlpane service.

4.1.7 Thrust Response

The system shall be capable of reaching 95% takeoff thrust from flight idle thrust within 5.0 seconds. The system shall be capable of reaching takeoff thrust from approach thrust (0.65 takeoff thrust) within 1.0 second for purposes of executing a go-around maneuver with the airplane.

The system shall be capable of reaching maximum reverse thrust from approach thrust within 1.5 seconds for purposes of arresting the landing roll on a 914.4 m (3000 ft) long runway without assist from the wheel brakes.

4.1.8 Emissions

The propulsion system shall incorporate exhaust pollution reduction technology to comply with QCSEE Program emission goals (levels as specified for 1979 aircraft by the EPA in the July 17, 1973 Federal Register, Vol. 38, No. 136, Part 2, Class T2).

4.1.9 Durability

The system durability shall be consistent with short-haul operations wherein the maximum utilization rates are 3000 flight hours per year and there are 700 - 1000 takeoff and reverse cycles per 1000 flight hours. Propulsion system condition indicators shall be consistent with the needs of performing maintenance on an on-condition basis rather than on a calendar or cumulative flight hour basis. This combined with confidence in component reliability, will permit maintenance costs per flight hour to be guaranteed.

4.1.10 Life and Duty Cycle

The engines shall be designed for a useful life of 36,000 hours over a 15 year period, based on the typical 402.3 km (250 mile) mission cycle shown in Table IV.

Cycle life shall be based on 48,000 mission cycles plus 1000 ground checkout cycles to full power. An aborted takeoff rate of 0.1 per 1000 takeoffs and an aborted landing rate of 0.1 go-around per 1000 flights should be assumed.

The above missions shall occur over the normal sea level and altitude ambient temperature distributions shown in Figures 14 and 15. The engine shall be capable of operation throughout the flight envelope shown in Figure 16. The corresponding Pt2-Tt2 envelope for a - 7.8° C (+18° F) day is provided in Figure 17, based upon inlet characteristics shown in Figure 18.

4.1.11 Flight Maneuvers

- The engine and its supports shall withstand without permanent deformation the conditions specified on Figure 19 (MIL-E-5007C except for precession rates). The calculated weight of the engine and engine-mounted nacelle components shall be increased by the specified weight allowed for all engine-mounted accessories.

- The engine and its supports shall withstand without failure static loads equivalent to 1.5 times the flight limit specified above for metal parts and 3.0 times for composite parts.

- At maximum allowable engine speed, the engine shall withstand without permanent deformation a gyroscopic moment imposed by a steady angular velocity of 1.0 radian per second in yaw, combined with a vertical load factor of ±1, for 15 seconds.

Table IV. Mission Cycle, Design.

Segment	Altitude km	Altitude Kft	Mach No.	% Thrust	Time, min	% Time
Start	0	0	0	3	0.5	1.1
Idle-Taxi	0	0	0	3 - 20	3.1	6.9
Takeoff	0	0	0	100%	1.22	2.7
Climb (1st Seg)	0 - 3.05	0 - 10	0.38	Max. Climb	5.0	11.1
(2nd Seg)	3.05 - 7.62	10 - 25	0.45	Max. Climb	5.0	11.1
Cruise	7.62	25	0.76	Max. Cruise	14.0	31.1
Descent	6.10	20	0.6 - C.3	Flight Idle	11.7	26.0
Approach	0.30 - 0	1 - 0	0.12	65% Max. F_N	1.3	2.9
Reverse Thrust	0	0	0.12 - 0	Max. Rev	0.08	0.2
Idle-Taxi	0	0	0	3 - 20	3.1	6.9
					45.0	100.0

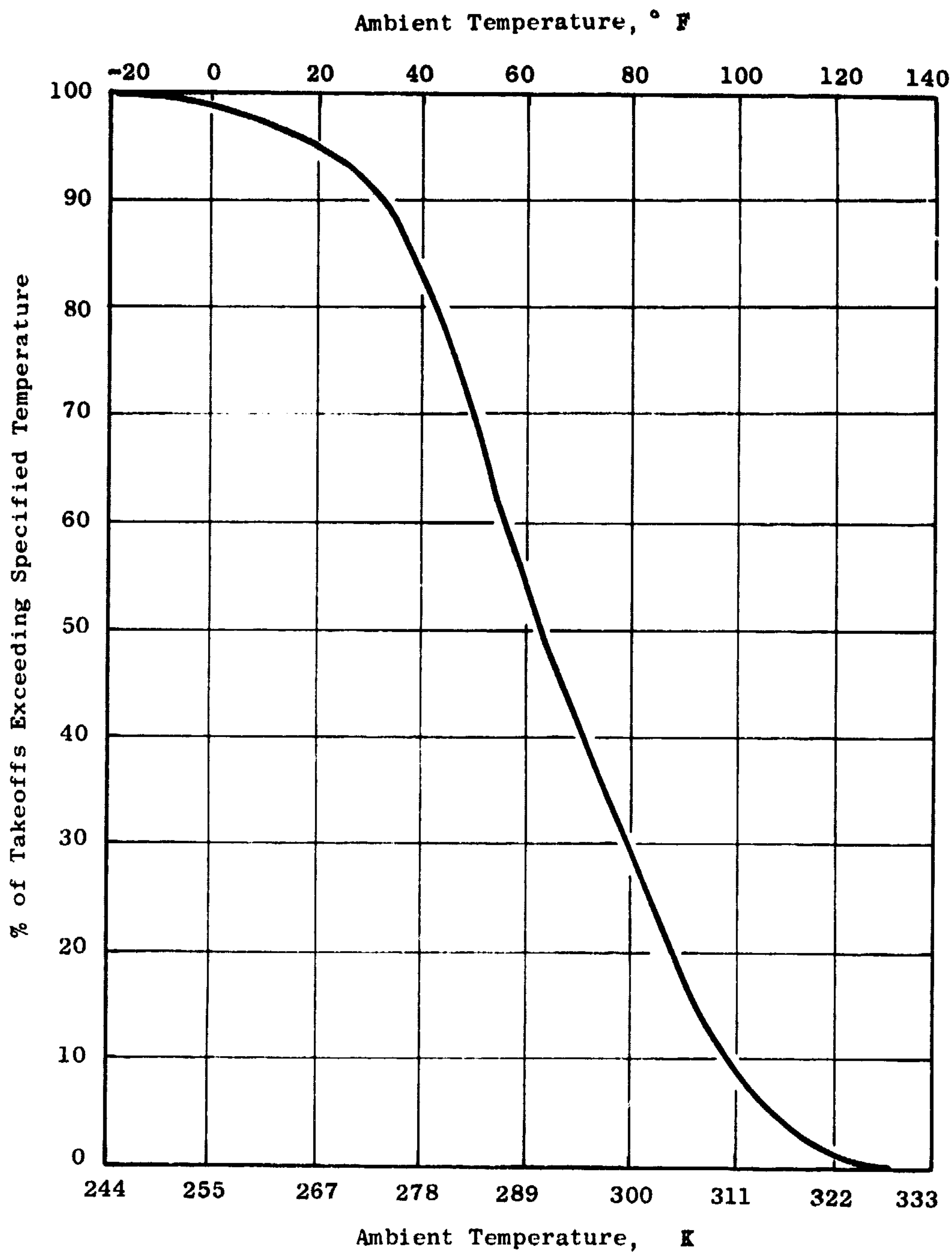


Figure 14. Takeoff Temperature Deviation.

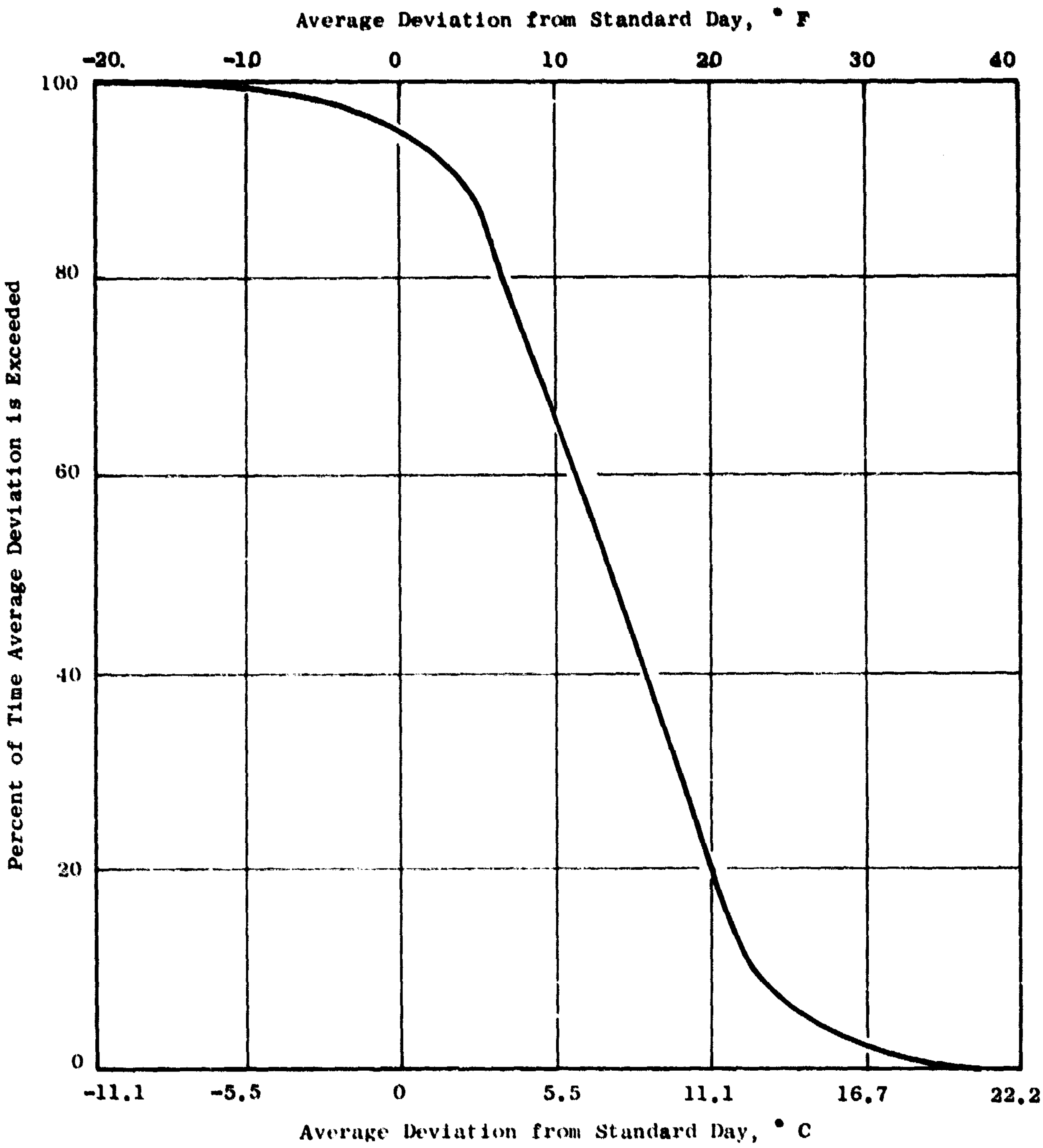


Figure 15. Flight Temperature Deviation.

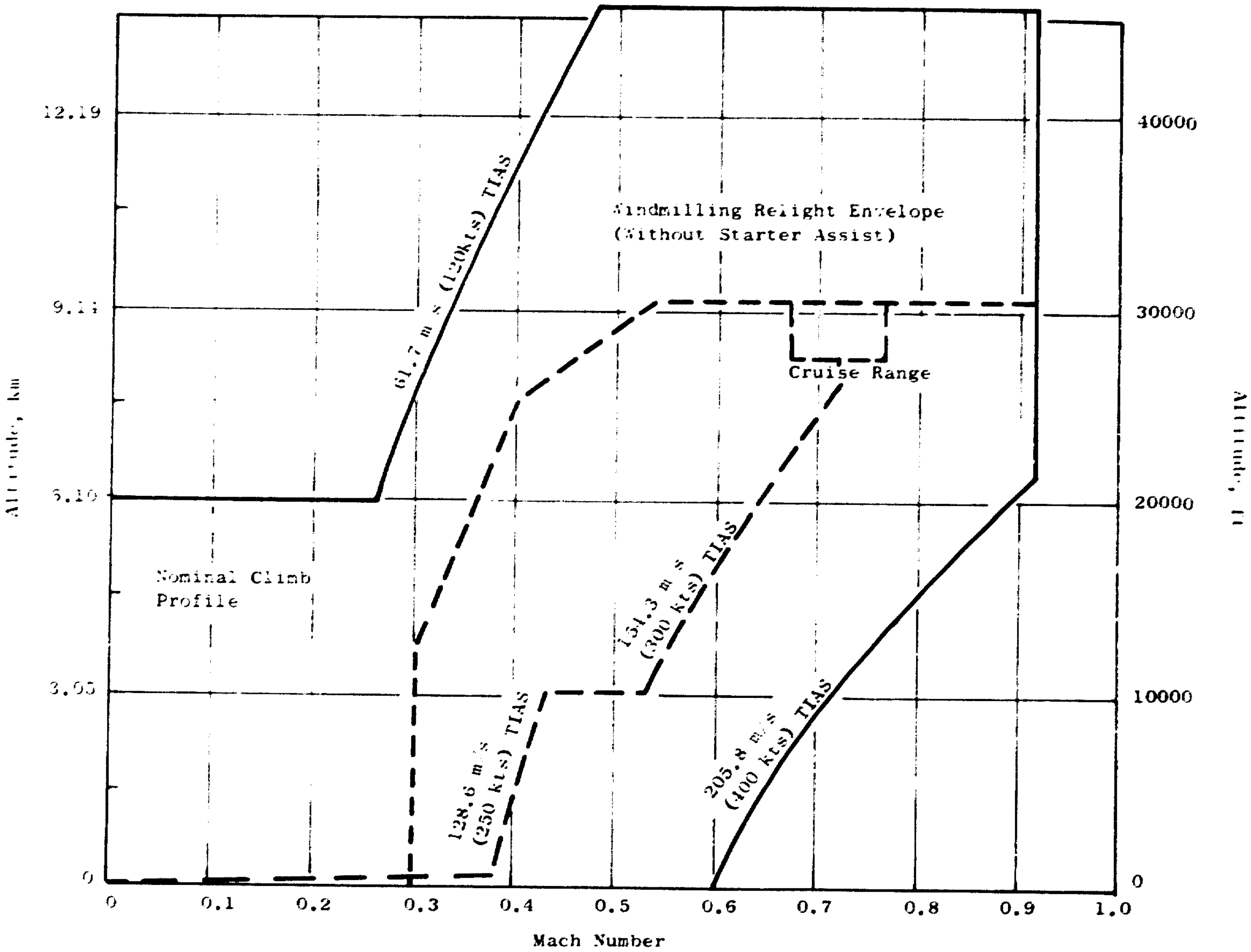


Figure 16. Operating Envelope.

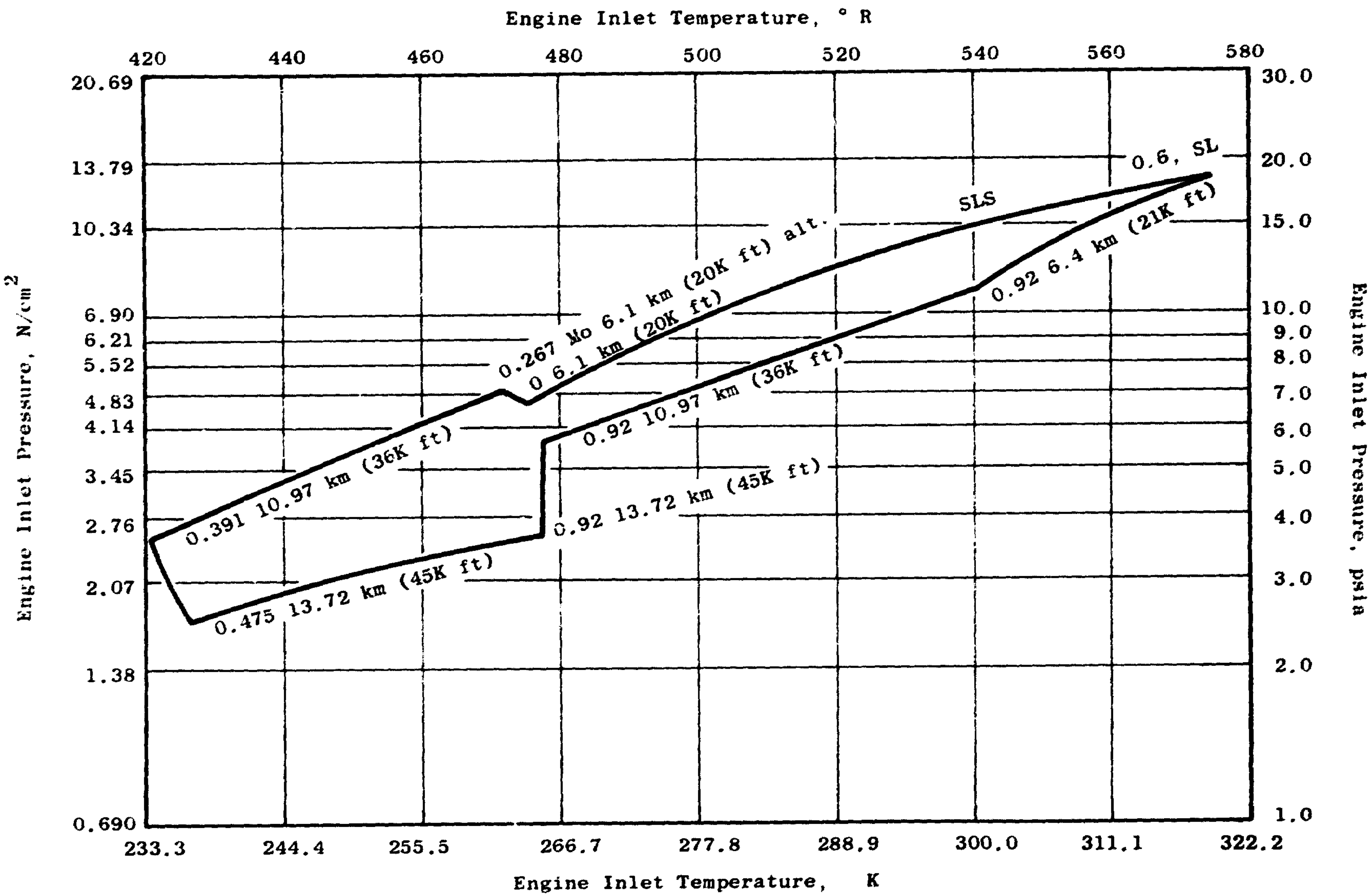


Figure 17. Inlet Temperature/Pressure Envelope.

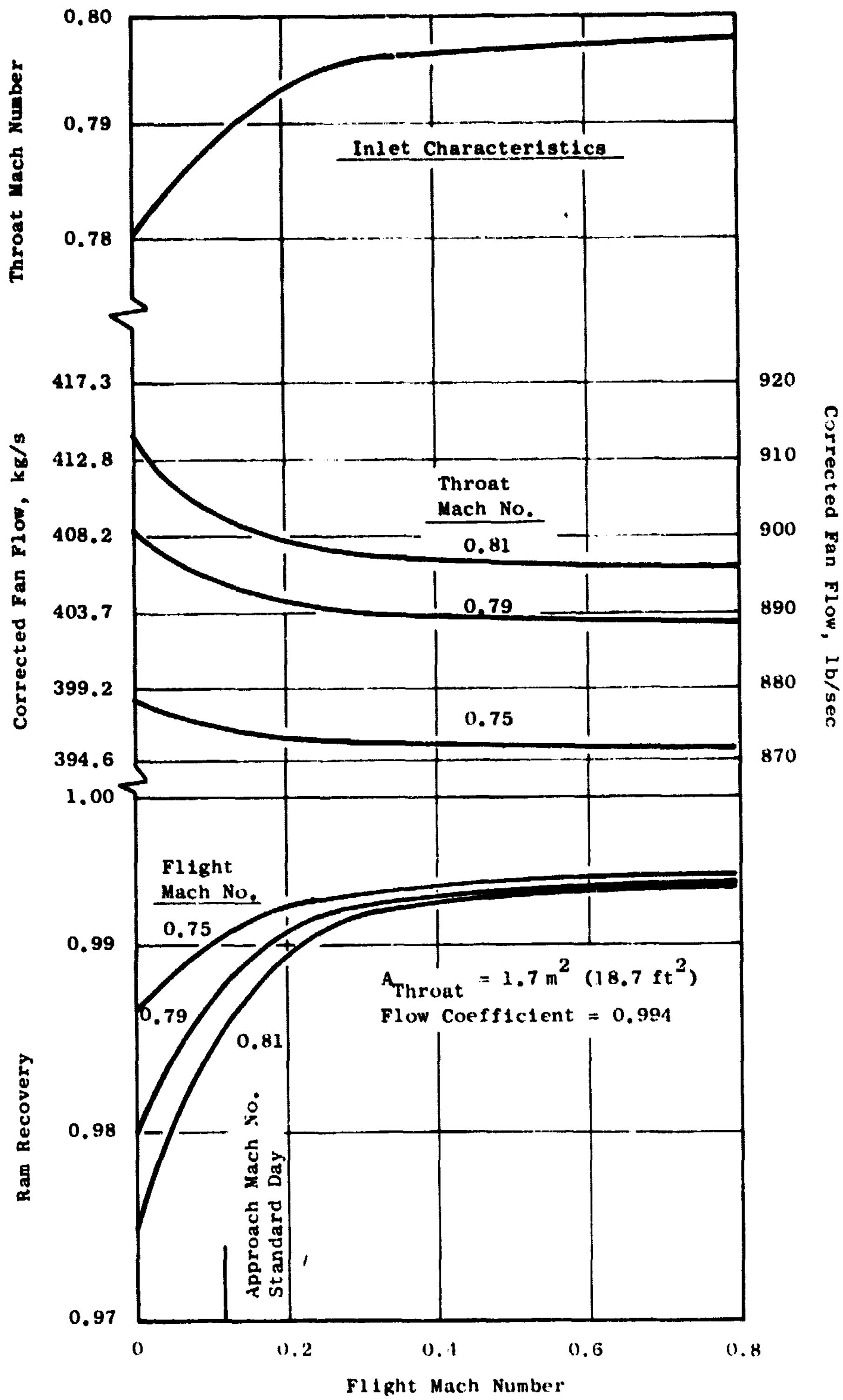


Figure 18. Inlet Characteristics.

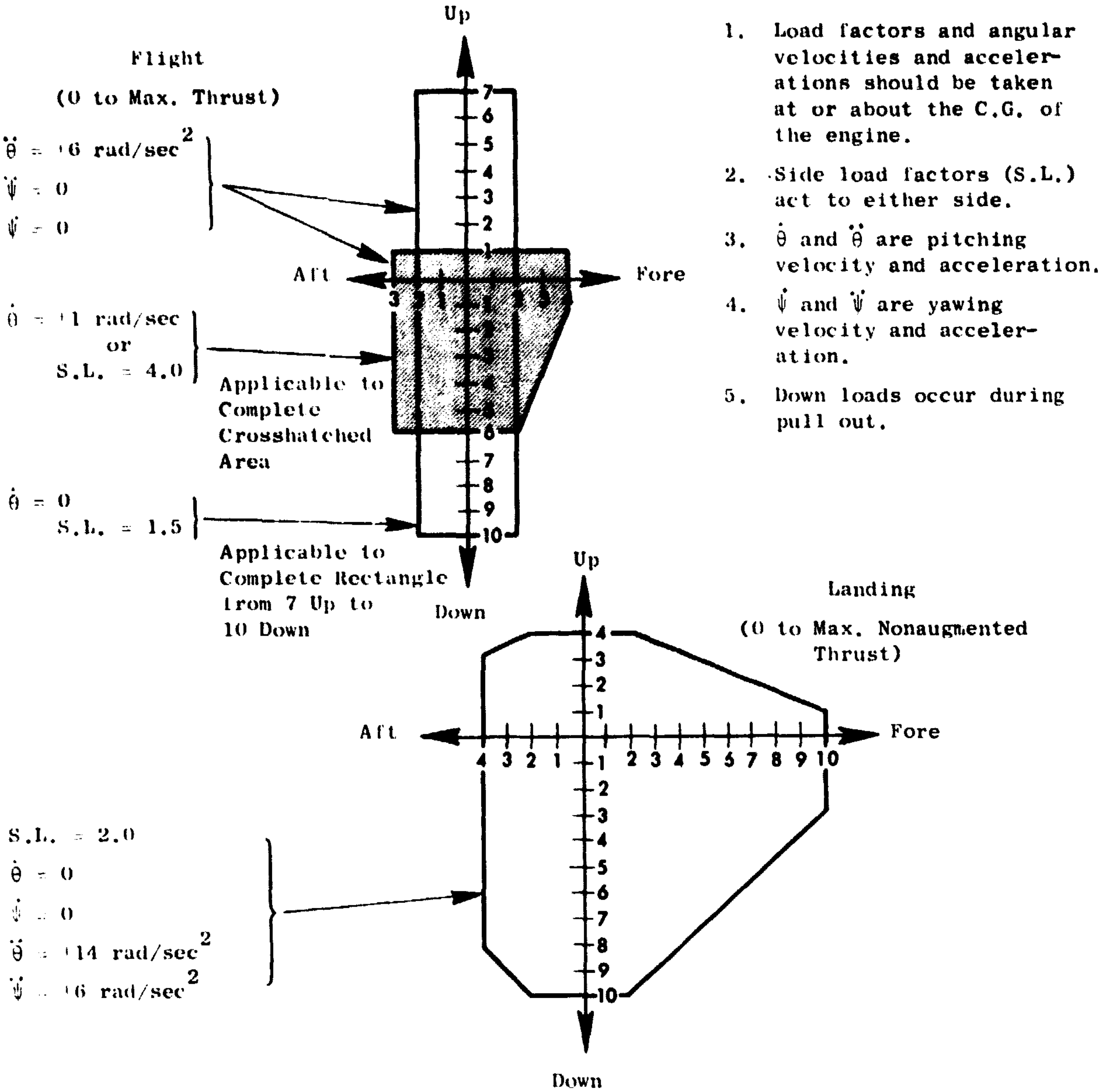


Figure 19. Maneuver Loads, Design.

- The engine shall be capable of withstanding loads caused by seizure of either rotor with deceleration from maximum rpm to zero rpm in one second.

- Composite parts shall be capable of withstanding unbalance loads caused by the loss of five adjacent composite fan blade airfoils at rated rpm. Metal parts shall be capable of withstanding unbalance loads caused by the loss of 2-1/2 adjacent composite fan blade airfoils at rated rpm.

4.1.12 Flight Attitudes

The engine shall be capable of operating within the range of flight attitudes shown in Figure 20.

4.1.13 Mounting and Installation

The engine shall be mounted from the wing box structure and cantilevered forward to locate the inlet in the free stream and the "D"-shaped exhaust nozzle exit at about 14% of the wing chord from the leading edge, see Figure 21. The aft nacelle section including the engine mounts, exhaust nozzle, and thrust reverser shall be integral with the wing such that the engine can be removed vertically downward as shown in Figure 22.

The mounting system shall consist of trunnions on the sides of the fan frame to react axial and vertical loads (vectors 1 and 2 in Figure 22). Side loads shall be reacted at the top of the fan frame (vector 3) and pitching moments by an aft vertical load mount at the top of the turbine frame (vector 4).

4.1.14 Lube Oil System

The lube oil capacity shall be compatible with utilizing the engines at least 10 hours per day without need to service the tank. Provisions are needed for indicating oil quantity to the flight crew and for filling the lube oil tank. The heat load from engine rotor bearings, accessory drive power train, and fan drive reduction gear should be transferred to the fuel system.

4.1.15 Accessory Drives

The engine-furnished accessories shall be mounted on the bottom of the engine for easy access and short service lines to the wing. Included in this accessory grouping is:

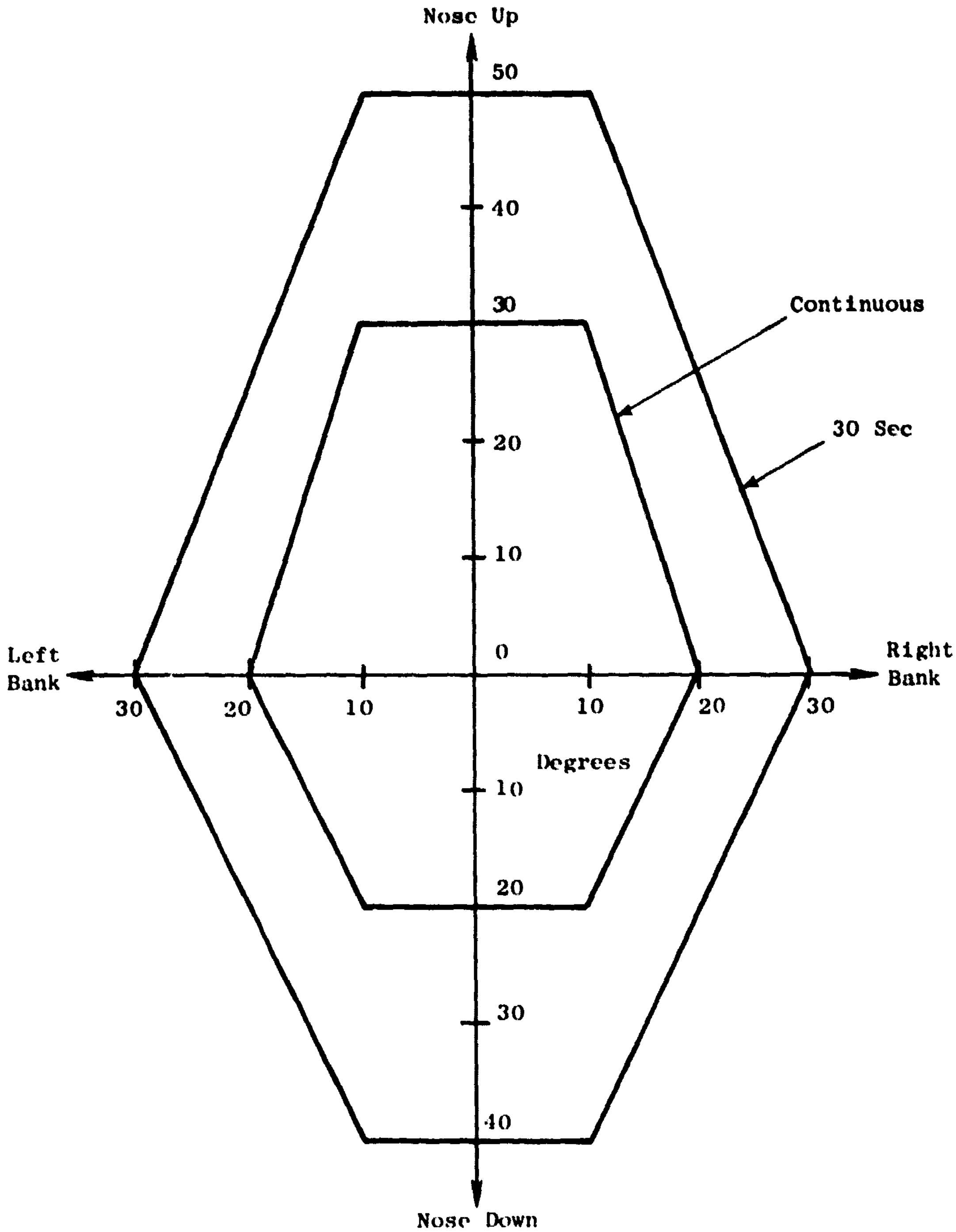


Figure 20. Design Attitudes.

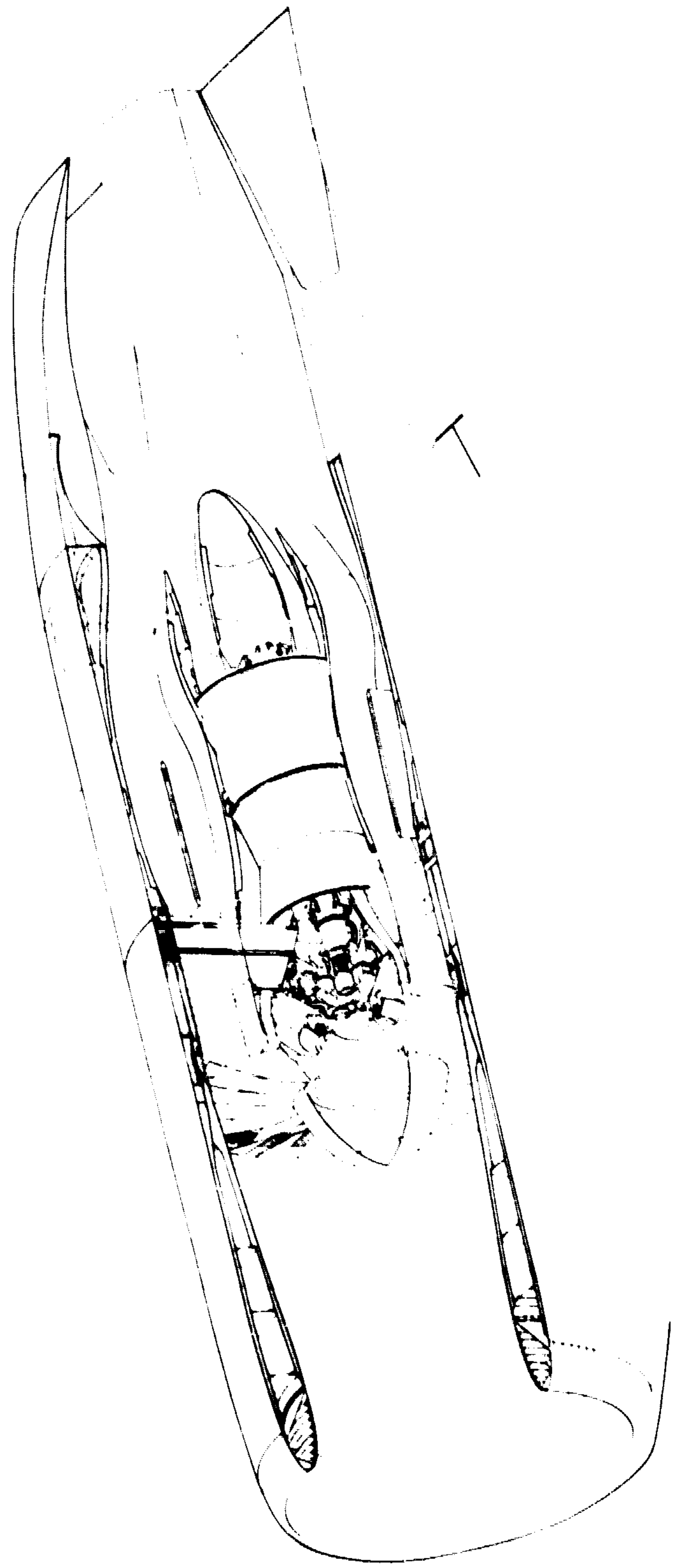


Figure 21. OTW Propulsion System Installation.

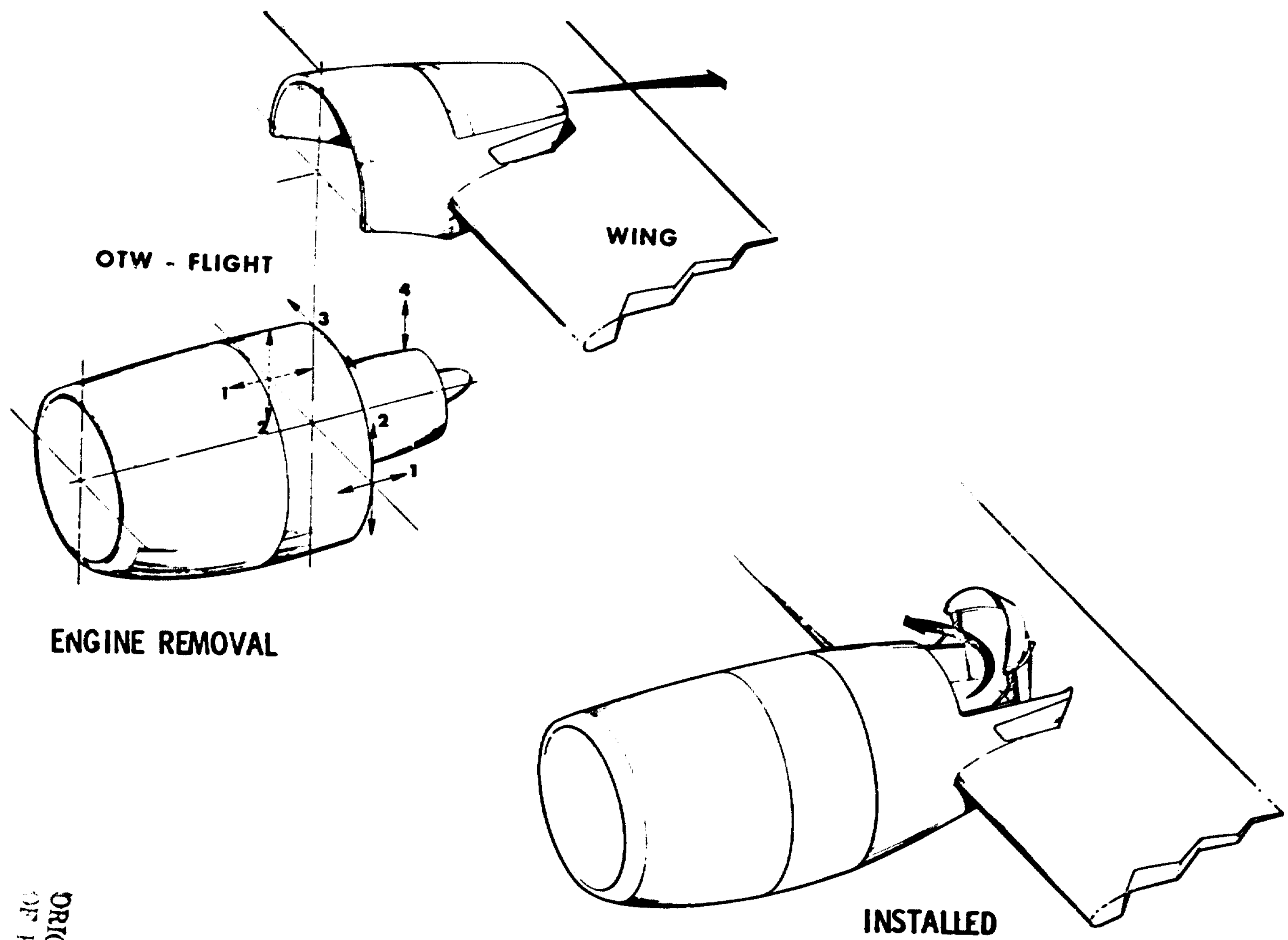


Figure 22. Flight Propulsion System Mounting and Removal.

- Fuel pump
- Digital control
- Fuel metering valve
- Lube and scavenge pumps
- Engine electric alternator
- Oil tank
- Ignition exciter

A removable accessory cover will provide access to these components.

Aircraft-furnished accessories will be located on a remote gearbox in the wing leading edge and driven by a power takeoff (PTO) shaft from the engine accessory gearbox. Included in these accessories are:

- Air turbine starter
- 60 kVA integrated drive generator
- Aircraft hydraulic pump

A separate accessory cover provides access to these components.

To remove the engine from the aircraft, the PTO shaft is removed, allowing the engine and its accessories to be lowered without disturbing the aircraft accessories.

4.1.16 Nacelle Ventilation and Cooling

Sufficient airflow must be provided through the core engine zone and the accessories section to purge the space of flammable vapors that might be present as the result of fuel or lube system component failure, and for cooling the components. Airflow rates that provide 6 - 10 air changes per minute in the affected spaces are typical of current practice and are considered necessary for this design.

For the core zone, it is envisioned that air from the fan discharge duct would be introduced through inlets near the front of the core cowl, sweep the length of the compressor, combustor, and turbine casings, and discharge from exits on the core cowl.

For the accessories zone, a ram scoop or scoops would introduce free stream air at the front of the zone, directing the flow at the engine lube oil tank and other selected accessory components. The air would flow rearward through the engine and airplane accessories sections, and exhaust to the atmosphere through an exit in the bottom of the nacelle. The airflow through this zone may be augmented by fan discharge bleed air if required for static operation.

4.1.17 Fire Protection

To implement the baseline airplane fire protection concept, the propulsion system must embody certain features to prevent the outbreak of fire and to contain an outbreak, in the event the prevention system is defeated, until the suppression system is brought into play.

The required features are, in addition to the flammable vapor purging function noted in discussion related to ventilation and cooling, the enclosing of all the fuel-carrying tubes in the core zone with fireproof shrouds that are vented to the atmosphere and fireproof construction at the boundaries of the core zone and accessories section.

The shrouded-tube concept serves to provide double-wall containment of the fuel. The vent feature affords a practical means of verifying the fuel-tight integrity of the primary (inner) tube and also of safely disposing of leakage from the primary, should that occur. The vent exit should be positioned to assure that leakage cannot possibly enter the engine air intake passages.

The fireproof construction is vital to the containment of a fire until the flight crew has become aware of the situation and taken action to isolate the affected zone by shutting off the various supplies of flammable fluids (fuel and hydraulic oil) and to suppress the fire by discharging the fire extinguisher system.

4.1.18 Drains

Provisions are required to facilitate collecting leakage that may develop with the accumulation of operating time, from accessory shaft seals, etc., and from various fluid leaks and spills that may occur in the accessory section as a result of component failure or maintenance actions, so that they can be safely discharged to the atmosphere. It is planned that this would be accomplished by routing the drain to the fan exhaust stream through the core nozzle shroud. Main engine seal drains would be routed overboard since leakage does not normally occur, but would provide indication of a malfunction.

4.1.19 Inlet Loads

The provisions for attaching the inlet duct to the fan frame must satisfy all of these loading conditions:

- Axial load stemming from the pressure distribution on diffuser, lip, and exterior cowl surfaces.
- The moments, shears, and axial loads related to the pressure distributions and inertia forces acting on the inlet during airplane maneuvers, gust encounters, and high sink-rate landings.
- The inertia forces associated with the forced vibration accompanying the loss of a group of adjacent fan blades at high fan rotor speeds.

4.1.20 Maintainability

Since the time required to replace engine components constitutes an important portion of the propulsion system maintenance load, the engine configuration should address minimizing this burden, thus reducing this component of airplane DOC.

The propulsion system should therefore be designed to meet the following goals:

- The engine should be capable of being trimmed on a test stand and not require additional trimming when installed on an aircraft.
- Access for inspection (including borescoping operations) and adjustments shall be accomplished without disturbing components or systems.
- Quick-opening doors shall be provided for access to the core to expedite turnaround and through-flight servicing actions.
- Borescope ports for inspecting the internal condition of the compressor, combustor, and turbines shall be suited to the performance of this task while the engine is in place on the wing.
- The engine shall be compatible with other nondestructive inspection/test techniques such as X-ray, zyglo, and radioisotope.
- The engine design should accommodate the forces related to maintenance personnel grasping and/or standing on those parts of the engine which could serve as hand holds or steps.

- Accessories and components requiring routine servicing actions must be accessible without need to remove or disassemble unrelated parts or systems.

- Fan blades must be replaceable without removal of the inlet duct. Fan duct surfaces must be durable enough to be compatible with meeting this goal.

- The tools required for line maintenance at way stations are limited to those reasonably expected to be part of an aircraft mechanics' kit.

- Engine hoist and ground handling equipment attach points must be accessible for the use intended.

- The engine assembly must be replaceable (on the wing) without disturbing the rigging of the throttle and fuel shutoff controls in the airplane.

For guidance in establishing goals, the accepted replacement times for selected (significant to QCSEE concept) engine components in current generation turbofan installations are noted below.

The time represents elapsed time needed to replace the designated component (see Table V), while the engine is in place on the aircraft, exclusive of the time required to gain access to the component (positioning ladders or work stands, gathering tools, obtaining the needed replacement part, opening cowls and access doors, the performing of functional testing, etc.). The tasks are performed by no more than three men having appropriate skills of average level.

On the basis of experience with current turbofan-powered airliners, it appears reasonable to use 240 minutes as a goal for the replacement time for the complete baseline engine change unit by a five-man crew.

4.1.21 Engine Control System

An engine control system based on an engine-dedicated, full-authority digital electronic controller is required for each engine. Each engine control system will provide selected automatic operating modes in response to thrust level and mode selection commands from the aircraft thrust control system. In addition, each engine control system will provide engine status data for crew display, recording, and on-board diagnostic purposes, over the entire airplane operating envelope. These engine status data will include selected failure indications and corrective action advisories generated within the engine control system.

Table V. Component Replacement Time.

Subsystem	Component	Minutes
Pneumatic	Intermediate bleed check valve	12
	CDP bleed control valve	15
	Inlet ice protection valve	10
Power Plant	Inlet duct	45
	Fan cowl half	20
	Core cowl half	20
Engine	Fan spinner	15
	Single fan blade	75
	Fan rotor assembly	250
	Combustor	65
	HP turbine	210
	LP turbine	210
	Fuel pump	60
	Fuel control	50
Exhaust	Core nozzle	80
	Nozzle plug	15
Lube Oil	Engine oil tank	20
	Lube pump - pressure	35
	Lube pump - scavenge	35
	Fuel/oil heat exchanger	25

4.2 INSTALLATION CONCEPT

Figure 23 shows the orientation of the propulsion system with airplane wing needed to achieve a high performance OTW externally blown flap (EBF) powered lift system.

4.2.1 Inlet Duct

The pod features a long, axisymmetric inlet duct with a relatively small throat section designed to induce a high inlet Mach number which serves to suppress the radiation of noise forward from the engine. The inlet highlight to throat diameter ratio of 1.21 adequately accommodates the expected inlet angle of attack that will be encountered in airplane operations. The inlet location, far-ahead of the wing, results in such modest upwash angles that there is no need to incorporate droop in the inlet. The inlet duct construction is a composite sandwich composed of Kevlar face sheets, epoxy bonded to aluminum alloy flexcore honeycomb. The leading edge section is all metal and is fitted with a hot air ice protection system.

4.2.2 Fan Module

The fan module is composed of a single-stage fan assembly featuring 28 fan blades of composite construction, supported in an all-composite fan frame which includes 33 integral stator vanes; a planetary reduction gear to drive the fan; the fan thrust bearing; and the power takeoff shaft which drives the engine accessories located on the bottom of the fan frame. The drive shaft is carried in the 6 o'clock position fan stator, which serves also as the leading edge section of the portion of the pylon that lies within the fan passage.

A belt of Kevlar felt is embedded in the fan frame on the exterior of the fan tip treatment to serve as a containment ring for the fan. No provisions are made to heat the spinner or fan stators for ice protection. Acoustic treatment is incorporated in the passages downstream of the fan, including the compressor inlet. The forward end of the fan frame is fitted with 16 rotary latches that are used to attach the inlet duct.

4.2.3 Gas Generator

The gas generator, or core engine module, is composed of a nine-stage, axial-flow compressor featuring variable-geometry inlet guide vanes and stators on the first three stages; a very short, high heat release, annular combustor and a single-stage, aircooled turbine. Ports are provided on the compressor casing for bleeding air from the 5th stage and from the compressor discharge.

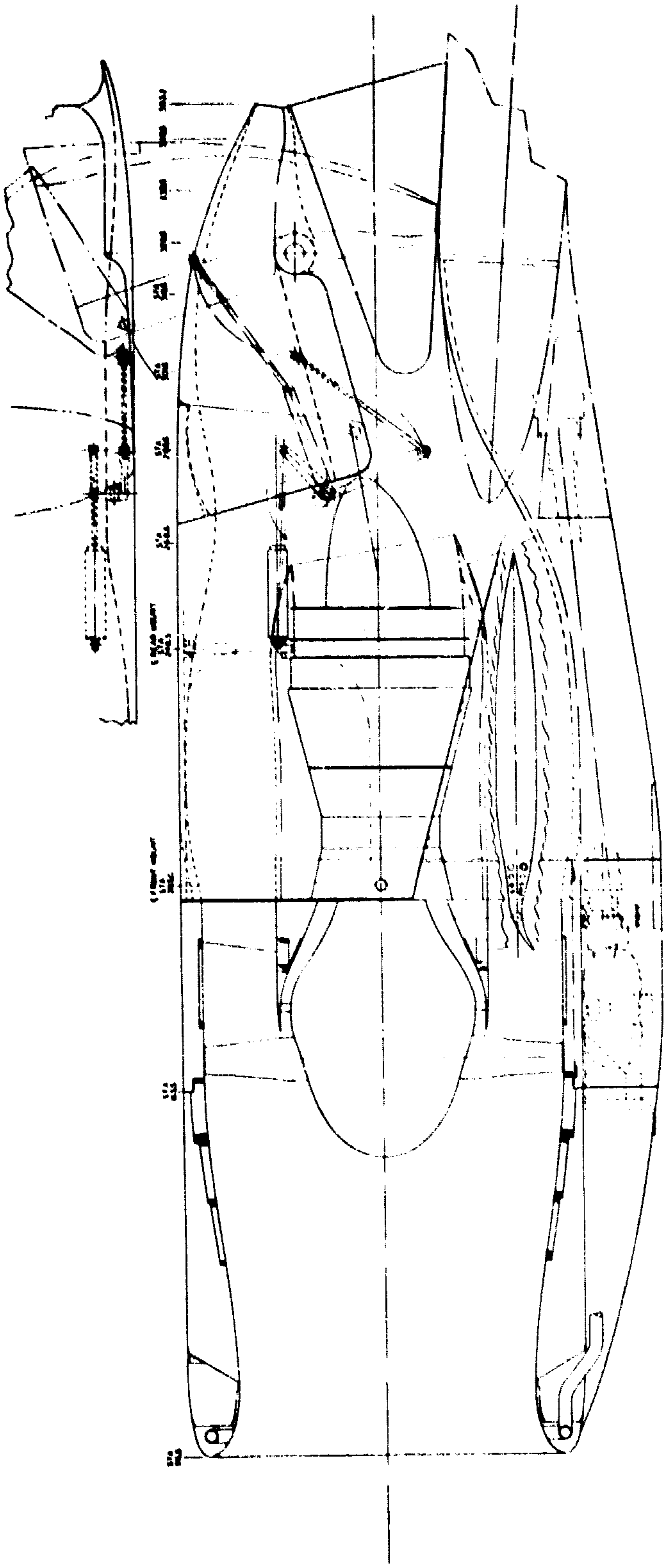


Figure 23. OTW Flight Propulsion System.

4.2.4 Power Turbine

The low pressure element of the engine turbine system drives the fan. The two-stage uncooled turbine is directly coupled to the sun gear in the planetary reduction gear assembly, and is supported by the turbine frame at the rear.

4.2.5 Reduction Gear

The reduction gear is an epicyclic star configuration having eight star gears rotating about fixed centers. Overall gear ratio is 2.0617. The entire gear assembly is located in the forward engine sump.

The reduction gear system effectively isolates the axial loads in the fan from those in the power turbine. As a result, the thrust load on the fan cannot offset the axial load on the power turbine and must be reacted solely by the fan thrust bearing. To compensate for the absence of the offsetting thrust from the fan, the power turbine is fitted with an air balance piston. This system, located at the rear of the turbine, utilizes air from the compressor discharge to apply a forward thrust force on the low pressure turbine rotor, offsetting the normal rearward load. This minimizes the thrust bearing load and permits the use of a smaller bearing.

4.2.6 Exhaust Nozzle/Thrust Reverser

The OTW engine incorporates a mixed-flow exhaust system in which the fan discharge air and turbine discharge gases partially mix, and exit through a common "D"-shaped exhaust nozzle. This nozzle is integral with the upper surface of the wing so that the exhaust stream spreads and turns over the wing flaps to provide powered lift. The required variation in exhaust nozzle effective area from 17,155 cm^2 (2659 $in.^2$) at takeoff to 14,316 cm^2 (2219 $in.^2$) at cruise is provided by hinged side doors, which enhance spreading in the takeoff (powered lift) regime. The core nozzle is canted upward 10 degrees to reduce wing upper surface heating from hot core exhaust scrubbing.

Thrust reversing is provided by rotating the upper surface of the exhaust nozzle to form a blocker and deflect the exhaust stream upward and forward. The forward portion of the blocker contains an articulated lip that completes turning of the exhaust stream to the desired exit angle. Side skirts, which rotate outward 45 degrees, are also employed for improved performance through reduced side spillage flow.

Both the area control doors and the thrust reverser blocker door are hydraulically actuated, using the aircraft hydraulic system for motive power. Since these components remain on the wing when the engine is removed, it is unnecessary to break hydraulic lines during an engine change. Position demand signals and feedback signals are provided electrically by the digital control.

4.3 NACELLE AERODYNAMICS

4.3.1 Inlet Design

The QCSEE inlet for the OTW propulsion system was designed for a throat Mach number of 0.79 and to operate at high angles of attack with sufficiently low engine-face distortion that adverse engine operation would not occur. The relatively high one-dimensional inlet throat Mach number was selected to achieve forward-radiating fan noise attenuation (as discussed in Section 7 on Nacelle Acoustics). The selected design throat Mach number provides sufficient margin from the inlet choking point for high internal performance, considering transient engine operational requirements, engine control tolerances, throat corrected flow variations due to aircraft operational effects, and inlet manufacturing tolerances. The high angles of attack resulted from anticipated STOL airplane angle-of-attack and crosswind conditions. The angle-of-attack condition defined by NASA requires satisfactory engine operation to 50 degrees at 41.2 m/sec (80 knots) air speed approaching the inlet. The NASA-defined crosswind requirement was for satisfactory engine operation with 18.0 m/sec (35 knots) wind speed from the side of the inlet (α_{cw} = 90 degrees). These levels were within the expected operating envelope of the YC-14 over-the-wing AMST aircraft design as provided by Boeing.

The initial design objective for the QCSEE inlet was to achieve the highest practical throat Mach number consistent with inlet/engine operation across the assumed aircraft operating envelope using a fixed-geometry inlet. As shown in Figure 24, based on representative subsonic inlet test results, a large degradation in inlet recovery is encountered at one-dimensional throat Mach numbers in excess of M_i = 0.82, due to effects of radial throat velocity gradients and boundary layer growth along the inlet lip. Consequently, to provide margin for effects such as engine-to-engine flow variation, flow variation due to operational effects on engine tolerances, and inlet-to-inlet throat area variation, a maximum practical design throat Mach number of 0.79 was selected.

The geometry for the QCSEE inlet design was established in scale model wind tunnel tests conducted at the NASA-Lewis 9 x 15 foot VSTOL Wind Tunnel. The scale model installed in the wind tunnel is shown in Figure 25. The diameter of the inlet duct at the simulated engine face was 30.5 cm (12 in.) which correspond to 16.9 percent of the full-scale inlet size. Aerodynamic performance of four axisymmetric inlet models with various leading-edge shapes was measured in the wind tunnel at angle-of-attack and crosswind conditions. The model geometries are summarized in Table VI. Results of the test program are summarized in Reference 8.

The inlet lip geometries that were tested were selected after review of NASA, GE, and Douglas available inlet data. The internal lip thicknesses tested corresponded to thickness ratios of R_{HL}/R_i = 1.17, 1.21, and 1.25. The internal lip shape was a 2:1 ellipse. Based on the available Douglas data, this range of lip thicknesses was sufficient to include

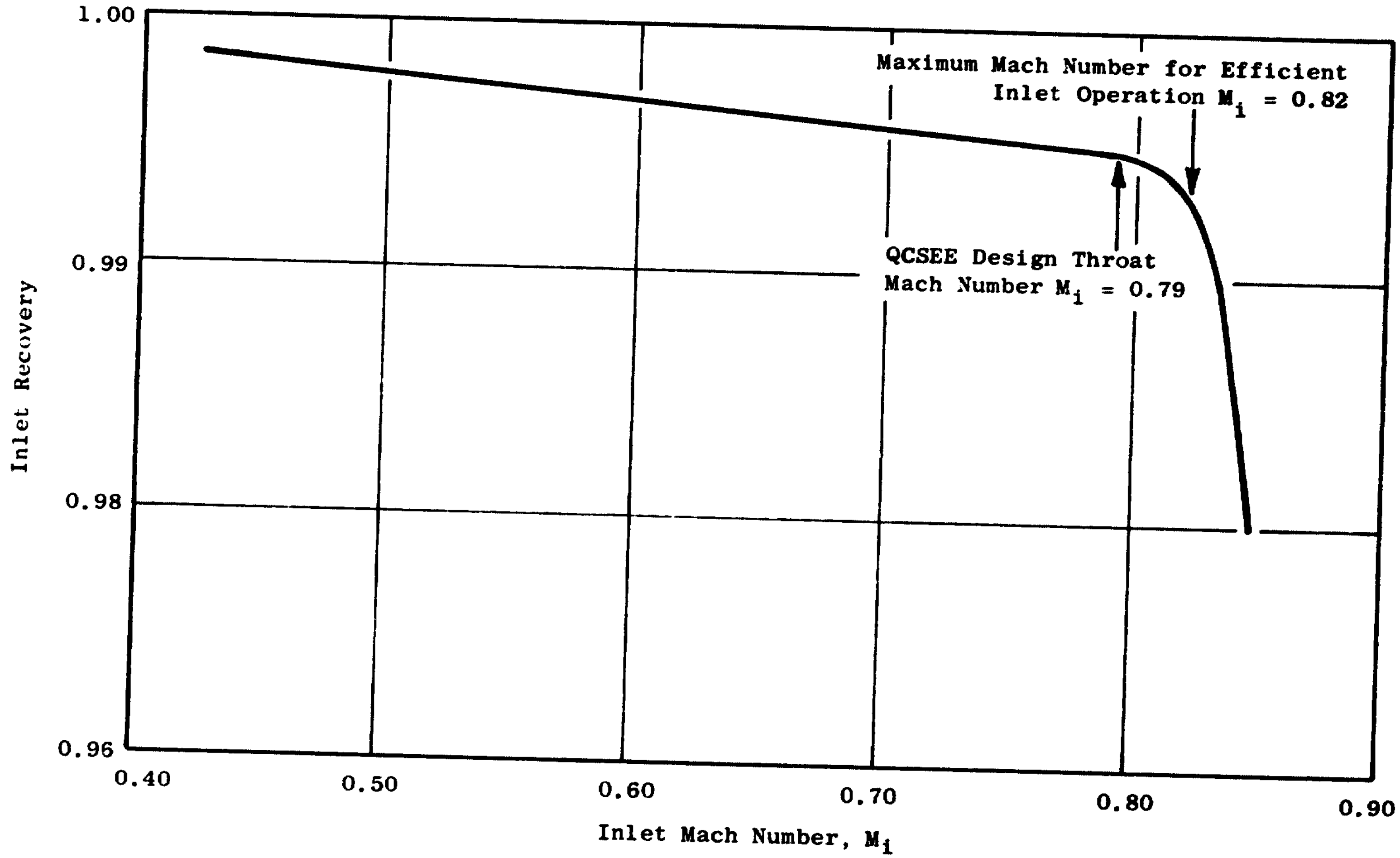


Figure 24. Inlet Throat Mach Number Selection.

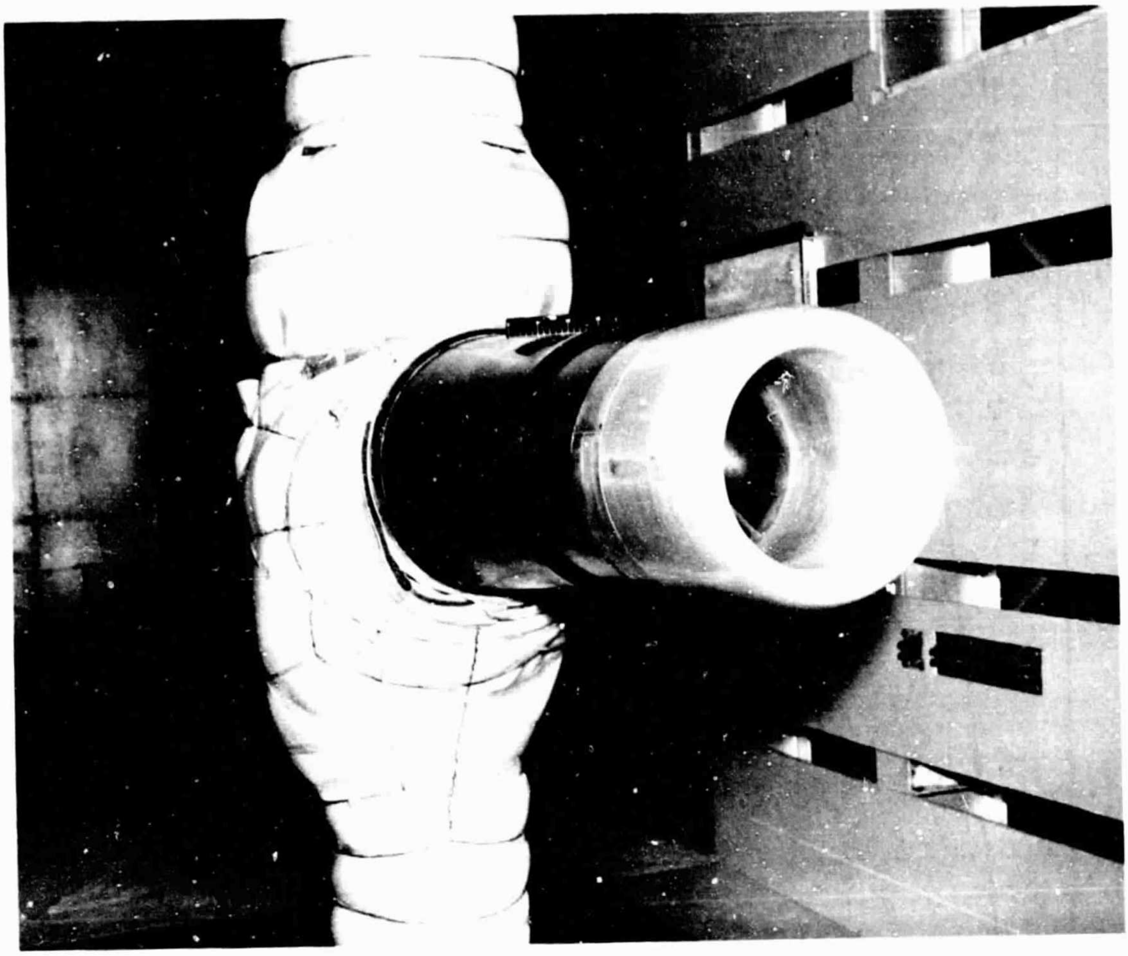
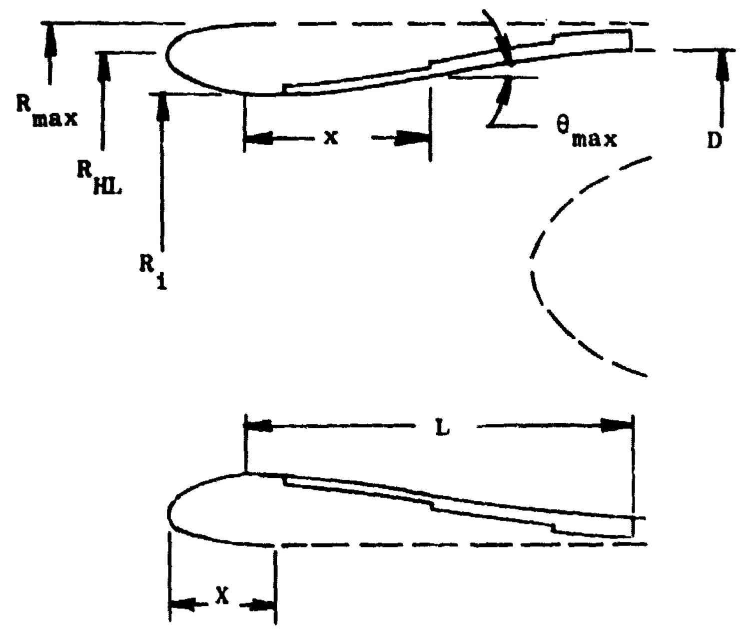


Figure 25. QCSEE 30.48 cm (12 in.) Inlet Model in NASA-Lewis 9 × 15 ft VSTOL Wind Tunnel.

Table VI. Model Test, Inlet Data.

Internal Lip		External Cowl			Diffuser			
R_{HL}/R_i	Shape	R_{HL}/R_{max}	X/D_{max}	Shape	L/D	$(x/L)\,\theta_{max}$	θ_{max}	θ_{eq}
1.17	2:1 ellipse	0.905	0.20	DAC-1*	0.83	0.50	8.7°	6.4°
1.21		0.905	0.20	DAC-1				
1.21		0.935	0.18	NACA-1				
1.25		0.905	0.20	DAC-1				



* DAC is Douglas Aircraft Company

that which would result in good inlet performance at angle-of-attack and crosswind conditions.

Two external-cowl geometries were tested to investigate the possibility that the internal flow might be significantly affected by the external-cowl shape at high angle-of-attack conditions. Both of these had an internal lip thickness ratio of 1.21. One geometry had a relatively blunt nose shape and was defined by Douglas (DAC-1) nondimensional cowl-shape coordinates, a cowl radius ratio of $R_{HL}/R_{max} = 0.905$, and a cowl-length ratio of $X/D_{max} = 0.20$. This geometry was also tested with internal-lip radius ratios of 1.17 and 1.25. The other geometry had a sharper nose shape defined by NACA-1 nondimensional cowl-shape coordinates, a cowl radius of $R_{HL}/R_{max} = 0.935$ and a cowl length ratio of $X/D_{max} = 0.18$.

One diffuser shape was used for all four inlet models. A diffuser length-to-diameter ratio (L/D) = 0.83 was sufficient for the required amount of full-scale-inlet acoustic treatment and also resulted in low aerodynamic-performance risk based on NASA theoretical studies and comparison with Douglas subsonic transport diffuser geometries. The diffuser inflection point at x/L = 0.5 was based on NASA theoretical studies.

Based on the model test results, the smallest internal lip that had the required angle-of-attack and crosswind capability was the $R_{HL}/R_i = 1.21$ lip (inlet number 2). This lip thickness was required for relatively low values of steady-state distortion $[P_{T_{max}} - P_{T_{max}}/P_{T_{avg}} < 0.10]$ for any value of airflow at the NASA angle-ot-attack and crosswind requirements. This is shown by Figures 26 and 27. The lowest lip thickness tested, with $R_{HL}/R_i = 1.17$, showed significantly higher distortion. Use of the larger lip thickness of $R_{HL}/R_i = 1.25$ would not have resulted in significantly better performance at high angle of attack conditions, but would have resulted in a larger external cowl maximum radius and, therefore, was not selected.

The test data for the four inlet configurations in terms of inlet recovery versus angle of attack are presented in Figure 28. As shown, two of the configurations exceeded the 50° angle of attack objectives and two failed to meet the objectives. On the basis of the test results, a 1.21 diameter ratio (R_{HL}/R_i) inlet with an external diameter ratio of 0.900 (R_{HL}/R_{max}) was selected as the best overall balanced design for angle of attack, crosswind, and cruise operation. The geometry of the selected inlet is presented in Figure 29.

4.3.2 "D"-Nozzle Design

The exhaust nozzle was designed to achieve the desired flow characteristics specified in the design requirements using the data base established by NASA-Langley (Reference 9). These data define the unique relationship of the aerodynamic turning and geometric nozzle and flap parameters. In addition, as the design evolved each design was presented to The Boeing

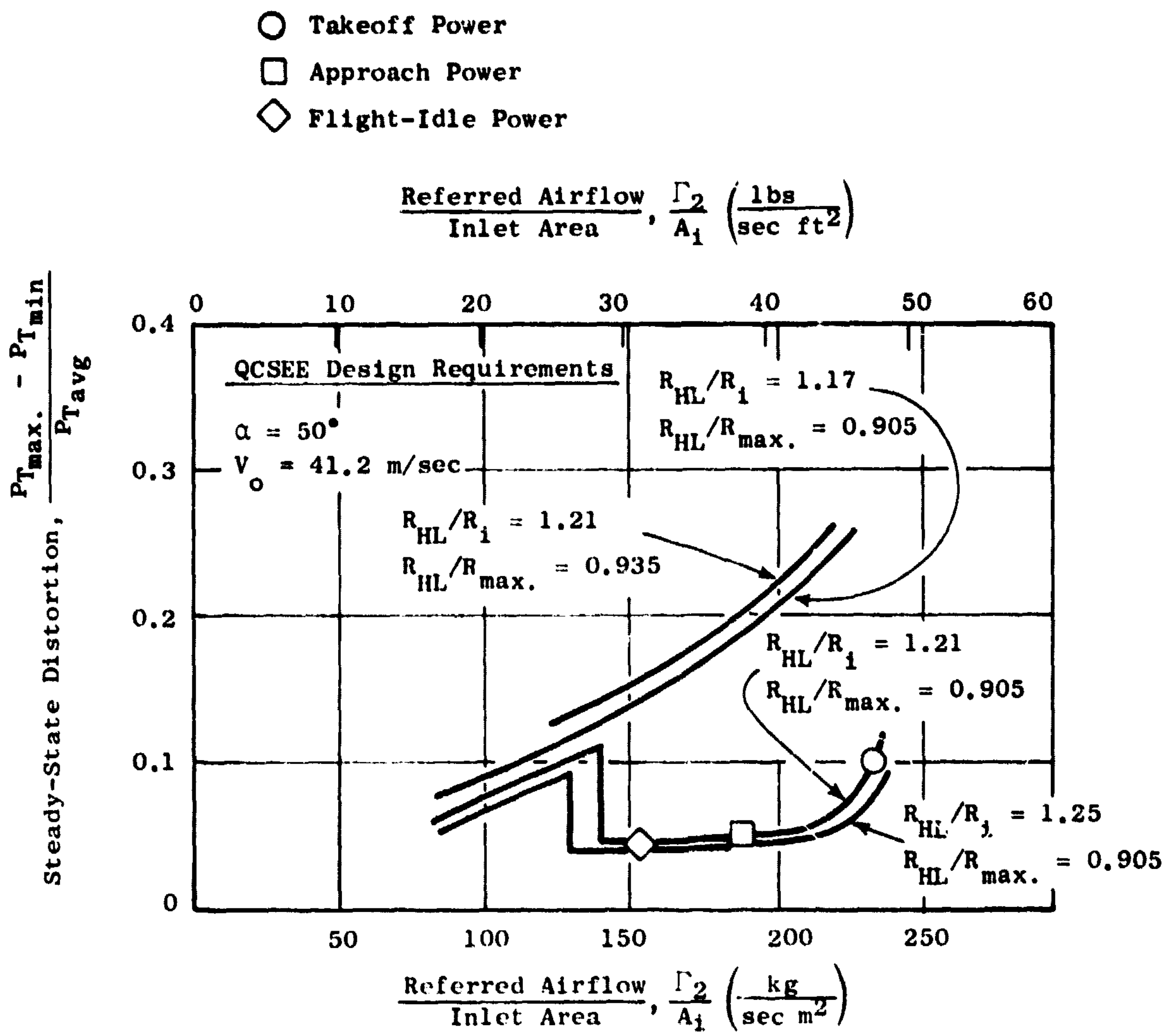


Figure 26. Inlet Angle of Attack Performance, Test Data α = 50°, V_o = 41.2 in./sec (80 kts).

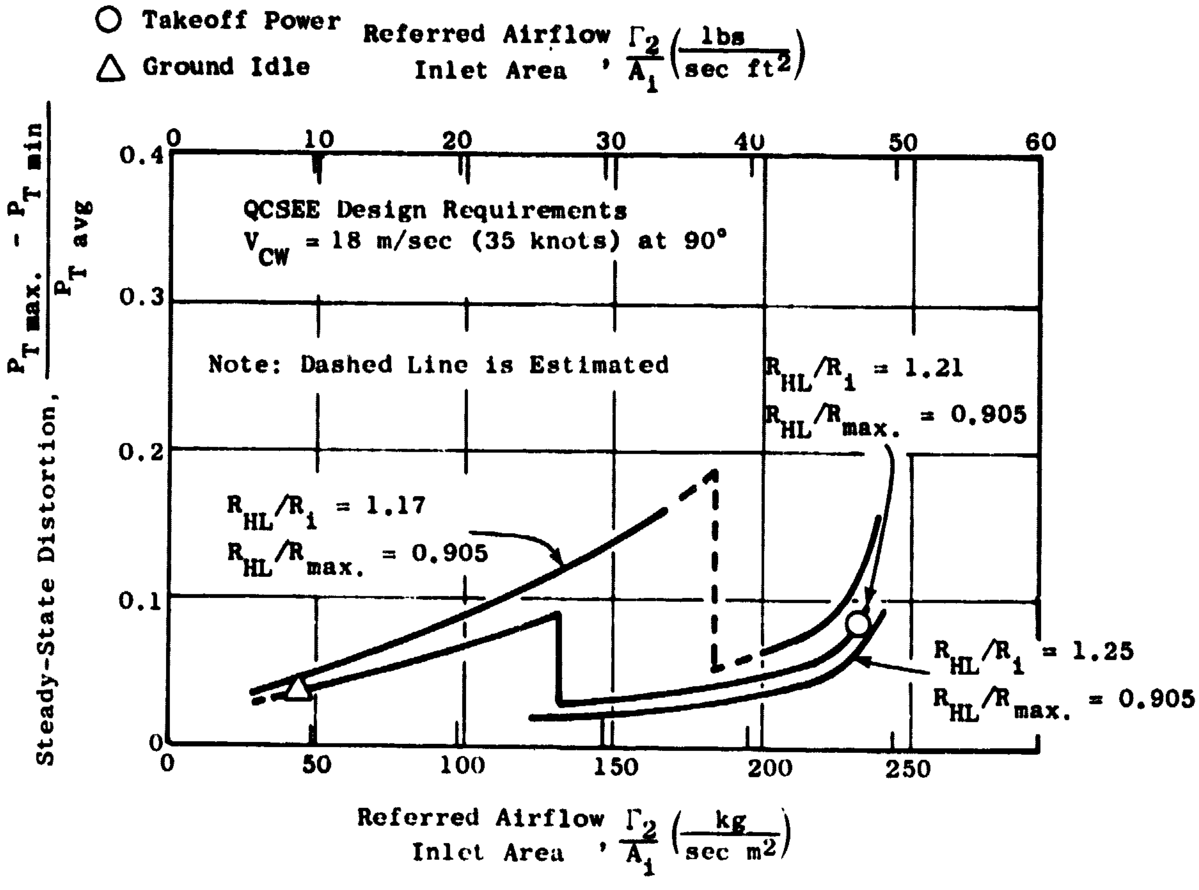


Figure 27. Inlet Crosswind Performance (Test Data).

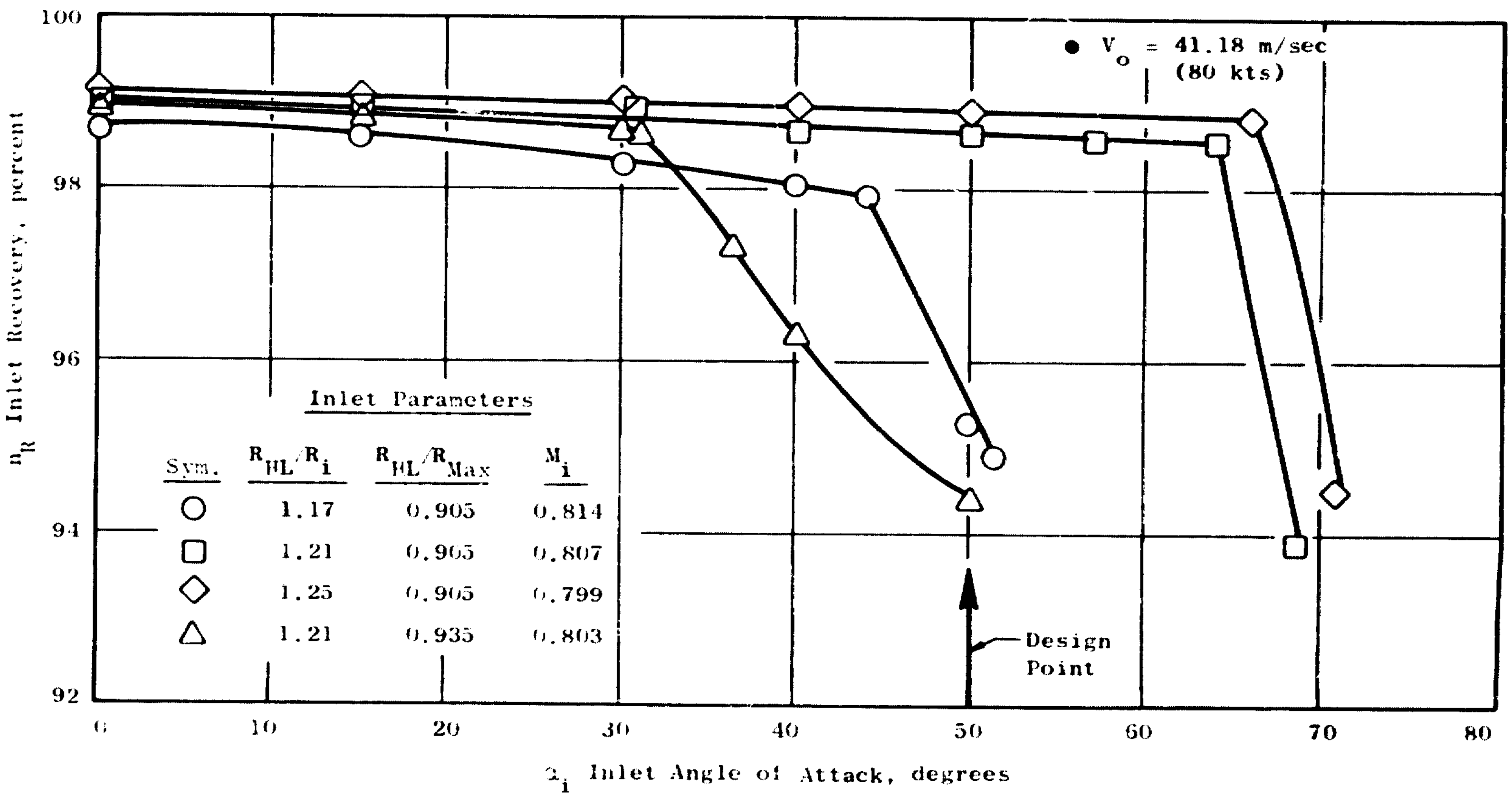


Figure 28. Inlet Performance Comparison Versus Angle of Attack.

Internal Lip is 2:1 Ellipse, $R_{HL}/R_i = 1.21$

External Cowl is DAC-1, $R_{HL}/R_{max.} = 0.90$, $X/D_{max.} = 0.22$

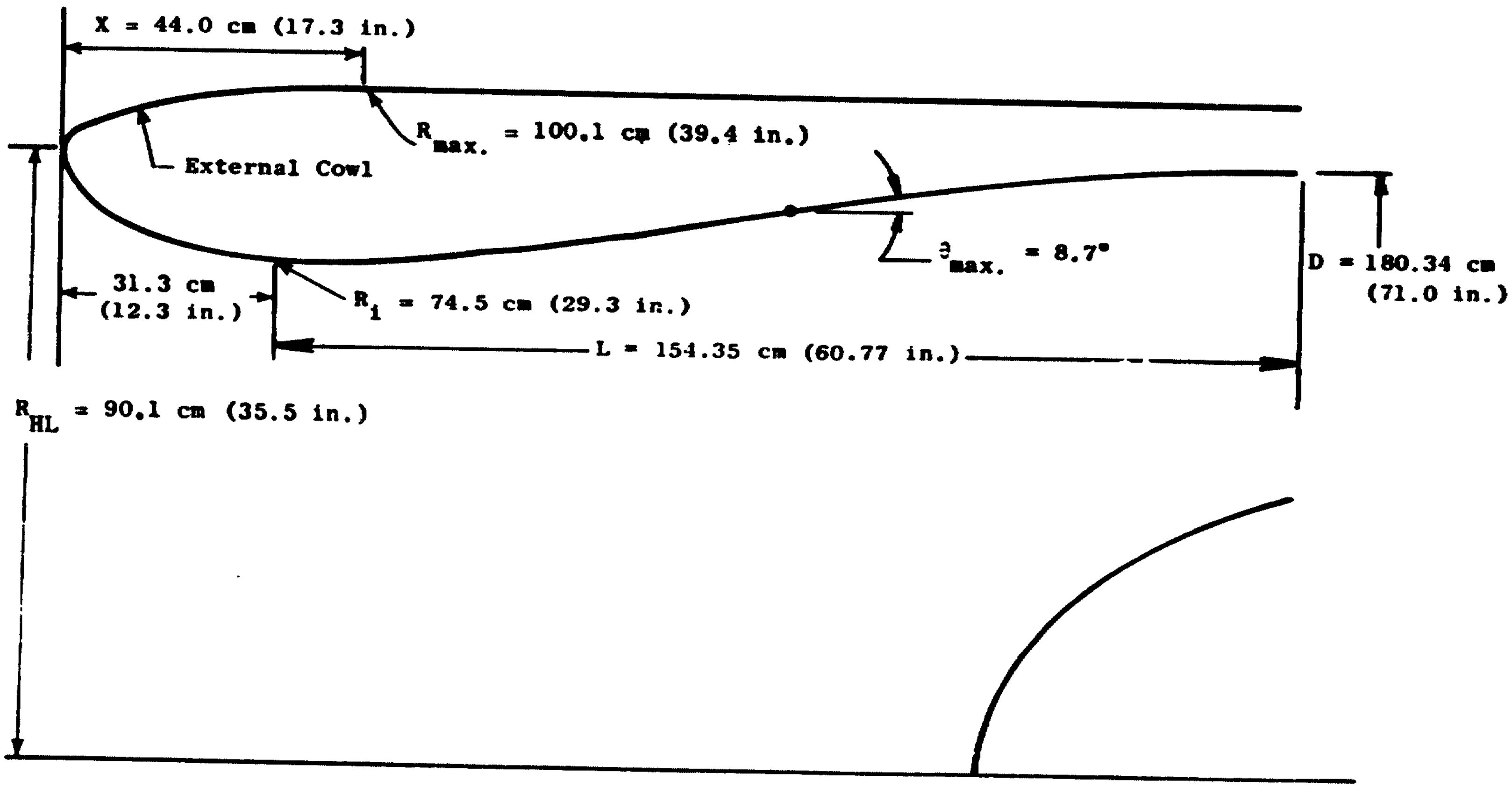


Figure 29. Inlet Lines ($M_i = 0.79$).

Company and generalized comments on each specific design were documented. The Boeing comments were utilized specifically to guide the external shaping and exit contour to ensure a proper blend between static and wind-on conditions, since no drag testing was planned.

The preliminary flowpath definition of the baseline QCSEE OTW exhaust system configuration is illustrated in Figure 30. The design incorporated side doors for the area variation and an upper target-type thrust reverser. The baseline design appeared to be aerodynamically and mechanically feasible but lacked a specific data base to launch any full-scale hardware design. The inadequate data base was due largely to the innovation of the two-sided door concept, which was unlike any other design available, and the difficulty in knowing what the exit area aerodynamic characteristics were in terms of flow coefficient, velocity coefficient, and flow spreading-flap flow turning. It was also desirable to confirm that the flap flow attachment achieved at static conditions would be maintained during wind-on conditions.

A joint GE, NASA-Lewis, NASA-Langley experimental program was established to obtain the required data base for the QCSEE nozzle and to confirm the performance relative to the design objectives. The results of this program defined the forward and reverse nozzle geometries to satisfy the design requirements previously stated and to confirm the wing flow turning characteristics under wind-on conditions. Test results are summarized in detail in NASA CR-134848 (Reference 2) and NASA CR-2792 (Reference 10).

The internal performance (flow coefficients, velocity coefficients) static test setup is shown in Figures 31 and 32. The model is shown with a bellmouth installed on a 14 cm (5.5 in.) diameter tip-turbine driven fan and round calibration nozzles at the exit. Shown on the test bench in Figure 32 is the QCSEE OTW baseline nozzle with side doors open to the takeoff position. This model was designed to allow ready interchangeability of the aft ducting. The model was pressure instrumented and thrust was measured using a two-component force balance mounted beneath the floor plane.

For wing/flap static turning performance, the OTW scale model and fan assembly were taken off the force balance and a wing, which was configured for the 60° approach flap setting, installed. The QCSEE nozzle was positioned over the wing upper surface and mounted to the test bench in a manner which did not make physical contact with the wing/force balance system, see Figure 33. With the propulsion simulator and QCSEE nozzle supplying airflow to the wing upper surface, the resultant wing lift and drag forces were measured on the force balance. These data were combined with nozzle-alone axial and vertical force data to obtain propulsion system wing/flap static turning angle and turning efficiency.

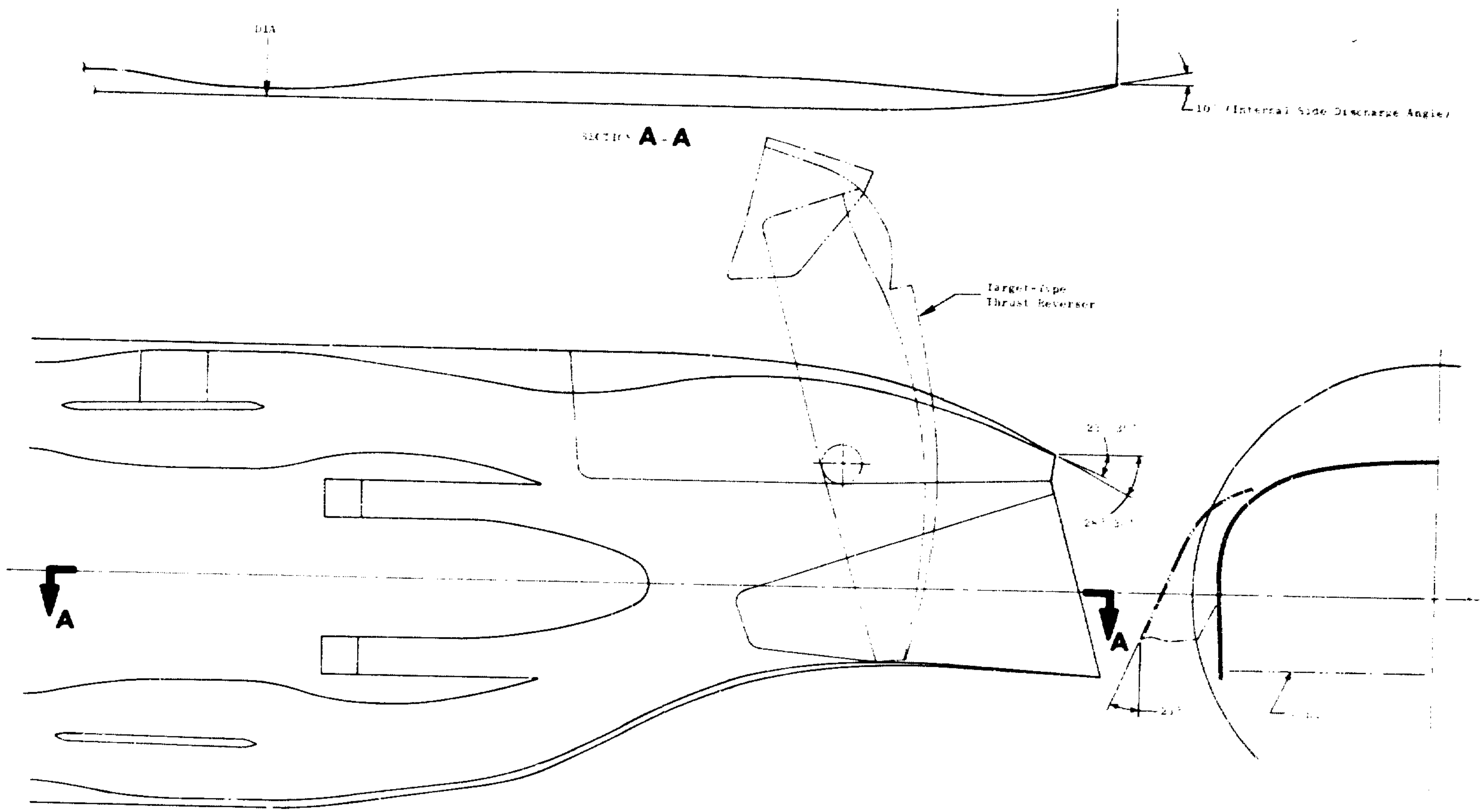


Figure 30. Preliminary OTW Baseline Propulsion System Design.

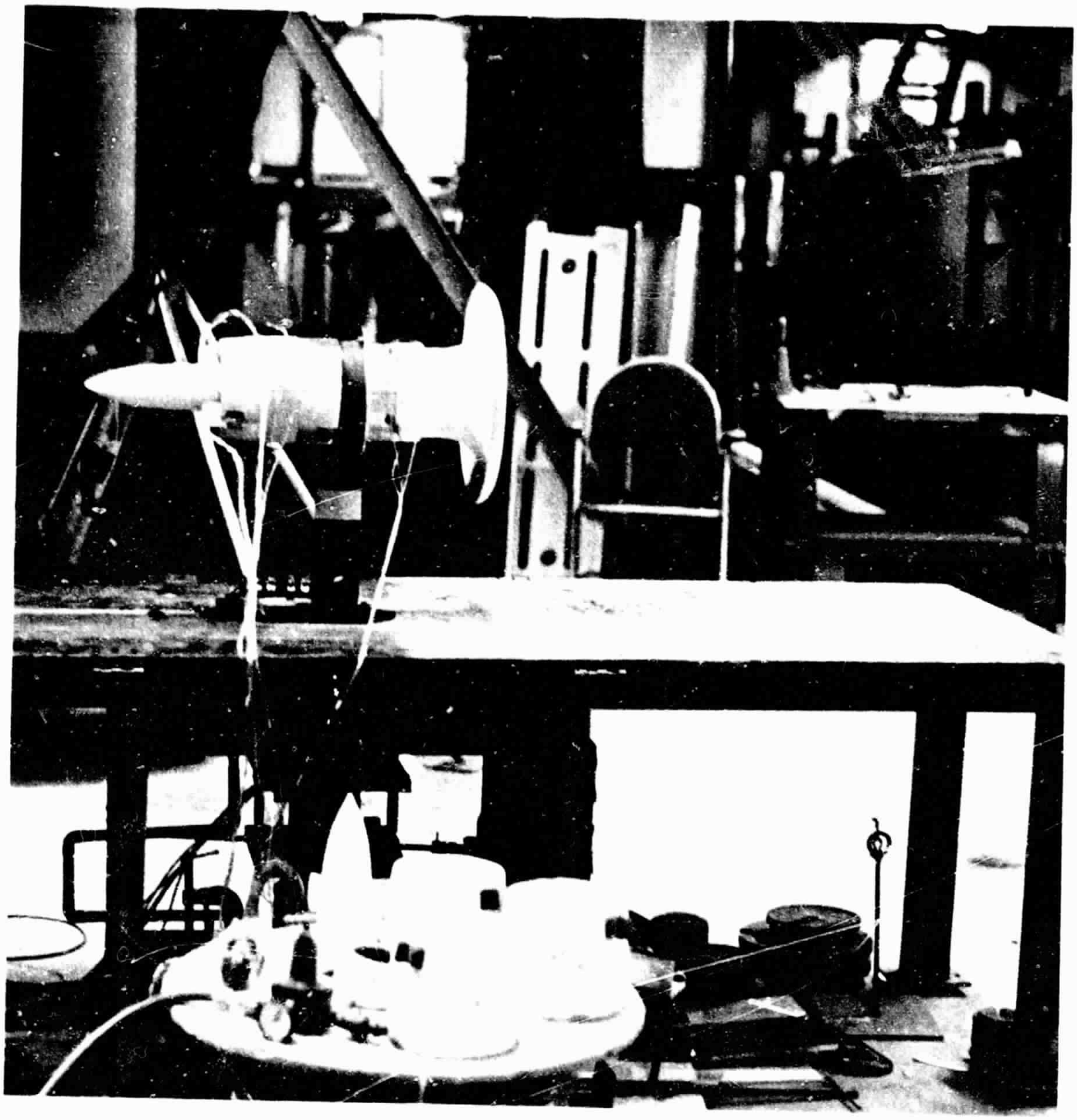


Figure 31. Nozzle Bench Test Setup.

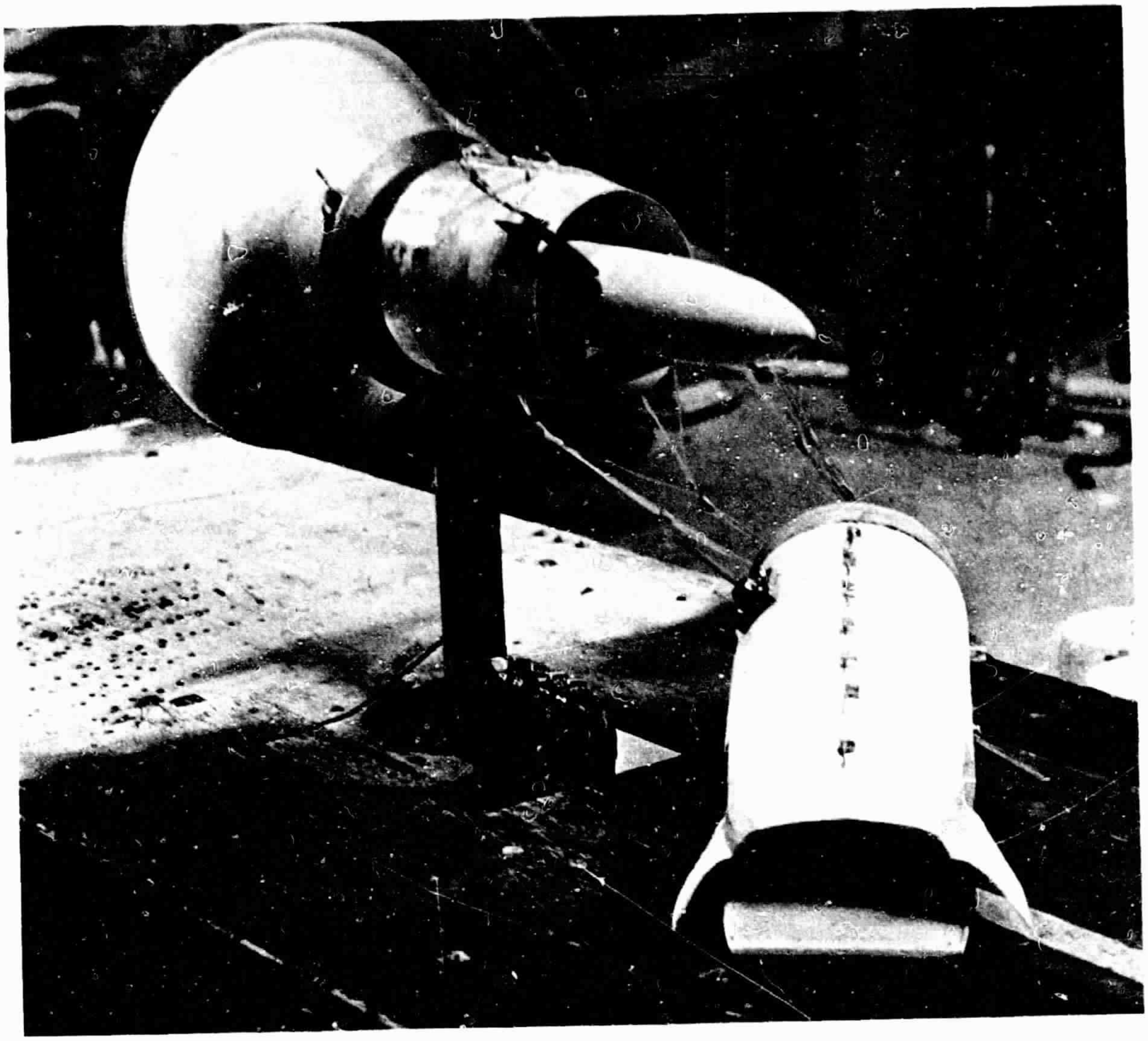


Figure 32. Nozzle Bench Test Setup.

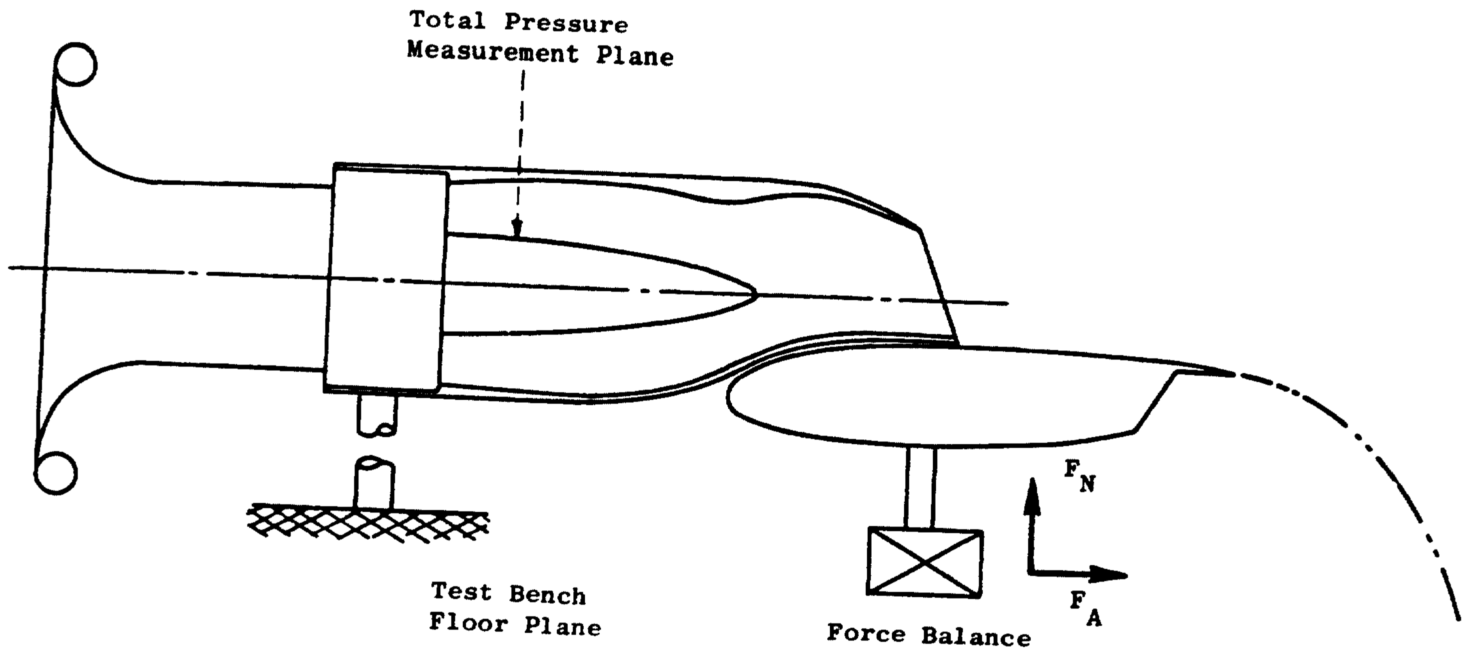


Figure 33. Wing Static Turning Test Configuration.

While the initial static turning test results of the baseline nozzle were very encouraging and in general agreement with the Langley data base (Reference 9), the measured static turning angle (48°) did not meet the QCSEE objective of 60°. Variations of the baseline nozzle were then made and tested in the program as shown in various views on Figures 34 through 36. The nozzle arrangements depicted on Figure 34, respectively, as the baseline, recontoured number 1, and recontoured number 2, reflected new technical insight into the flow turning mechanism and the direction to proceed for improving static turning with some attention to the impact on external drag. Both of the recontoured nozzle configurations included changes in nozzle floor and roof flowlines to provide more downward direction of exhaust flow (steeper boattail and floor angles) for increased flow spreading on the wing and, therefore, better flow turning around the flap. Figure 34 shows the magnitude of the change in roof and floor angles, while Figure 35 indicates the extent of flowpath change along the vertical centerline and also the lateral extent of change in roof line as shown from the cross sections at Station 348. The nozzle exit plane was not changed on any of the modifications. Two side door arrangements were included in the experimental evaluation, a small door with angle of 60° and large door with angle of 25° described in Figure 36.

The static turning data for RC-1 and baseline nozzles is shown in Figure 37. The recontoured RC-1 nozzle produced significantly better turning data than the baseline, with relatively insignificant difference between the large and small door designs. RC-2 data indicated the same degree of static turning, but these results are not shown because of suspected force balance measurement errors.

The recontoured RC-1 nozzle performance with large nozzle side doors was considered very satisfactory, meeting all turning objectives, and all consideration for alternate concepts was dropped at this point. Only the wind-on test was required to ensure that the flow attachment was not affected by freestream effects.

Figures 38 and 39 present flow coefficient data for the baseline and RC-1 nozzles. Both nozzles show essentially the same area change between takeoff and cruise effective areas. The cruise-to-takeoff area change measured is considered acceptable for the QCSEE OTW demonstrator engine cycle needs.

Nozzle velocity coefficient data are shown on Figures 40 and 41, for the baseline and RC-1 nozzle configurations. These velocity coefficients are defined as the resultant velocity (determined from vertical and axial force balance readings and measured airflow) divided by the calculated ideal velocity (function of pressure ratio). The results show that RC-1 and baseline takeoff velocity coefficients are essentially the same ($\sim$0.910 to 0.915).

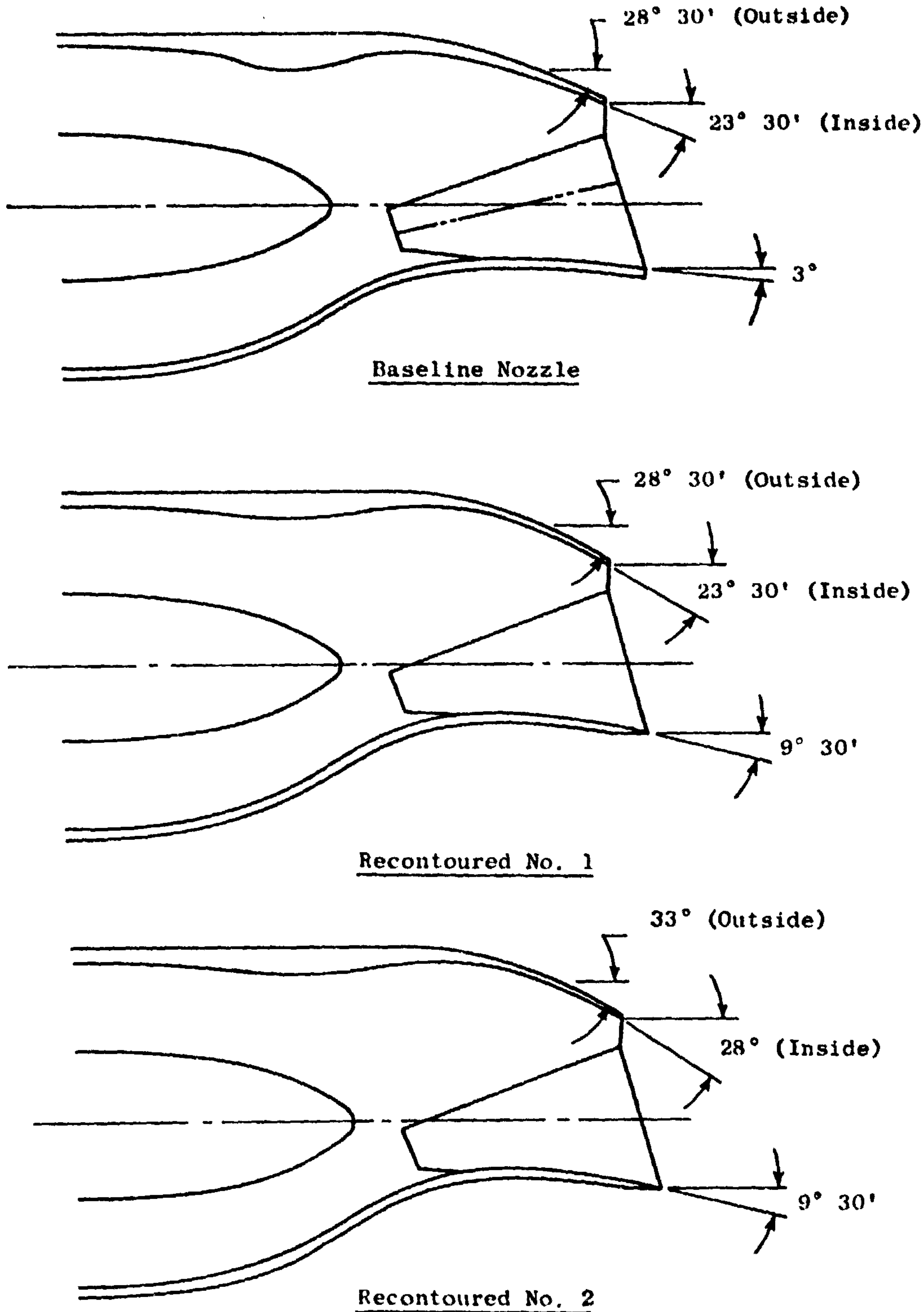


Figure 34. Baseline Nozzle Variation.

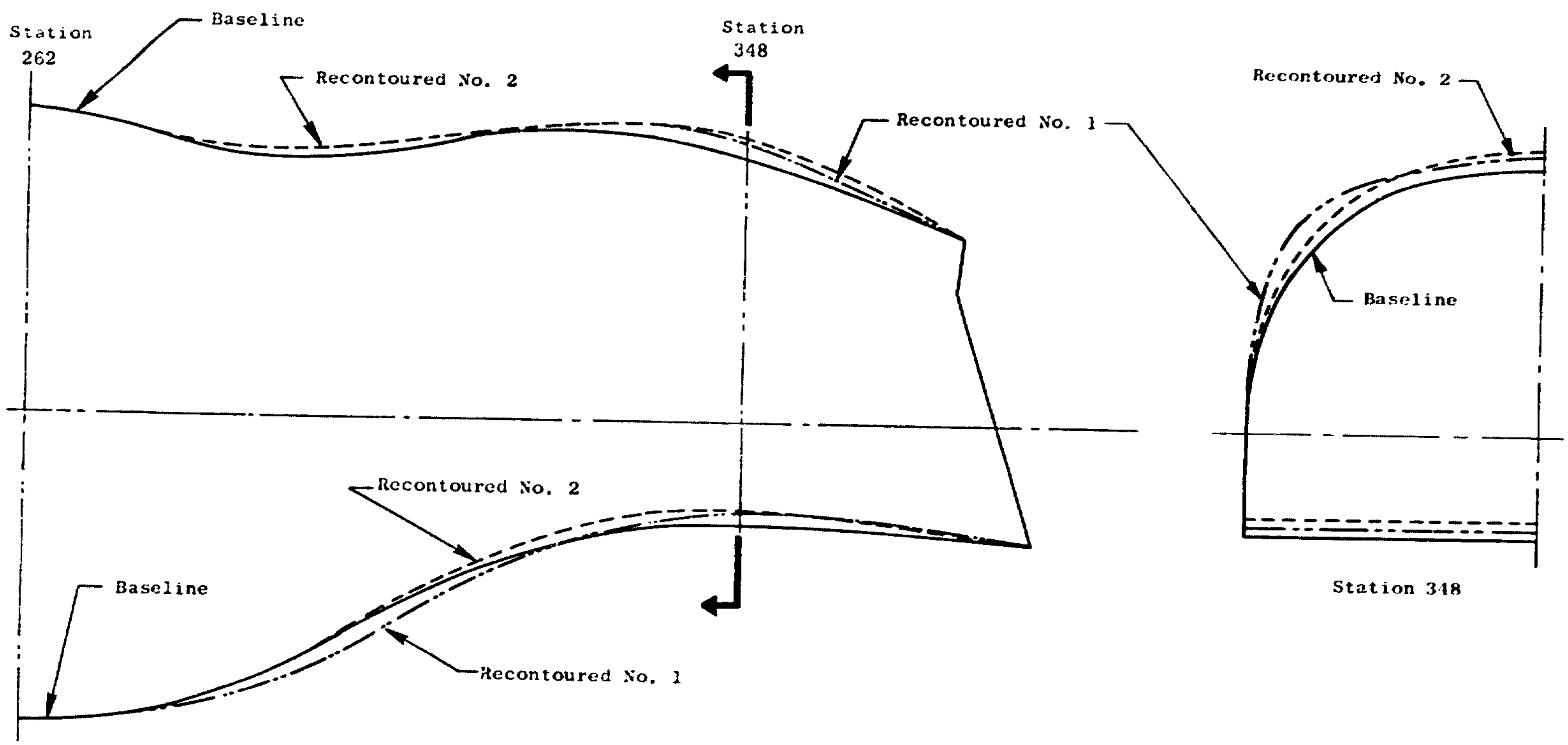


Figure 35. Nozzle Internal Flowpaths.

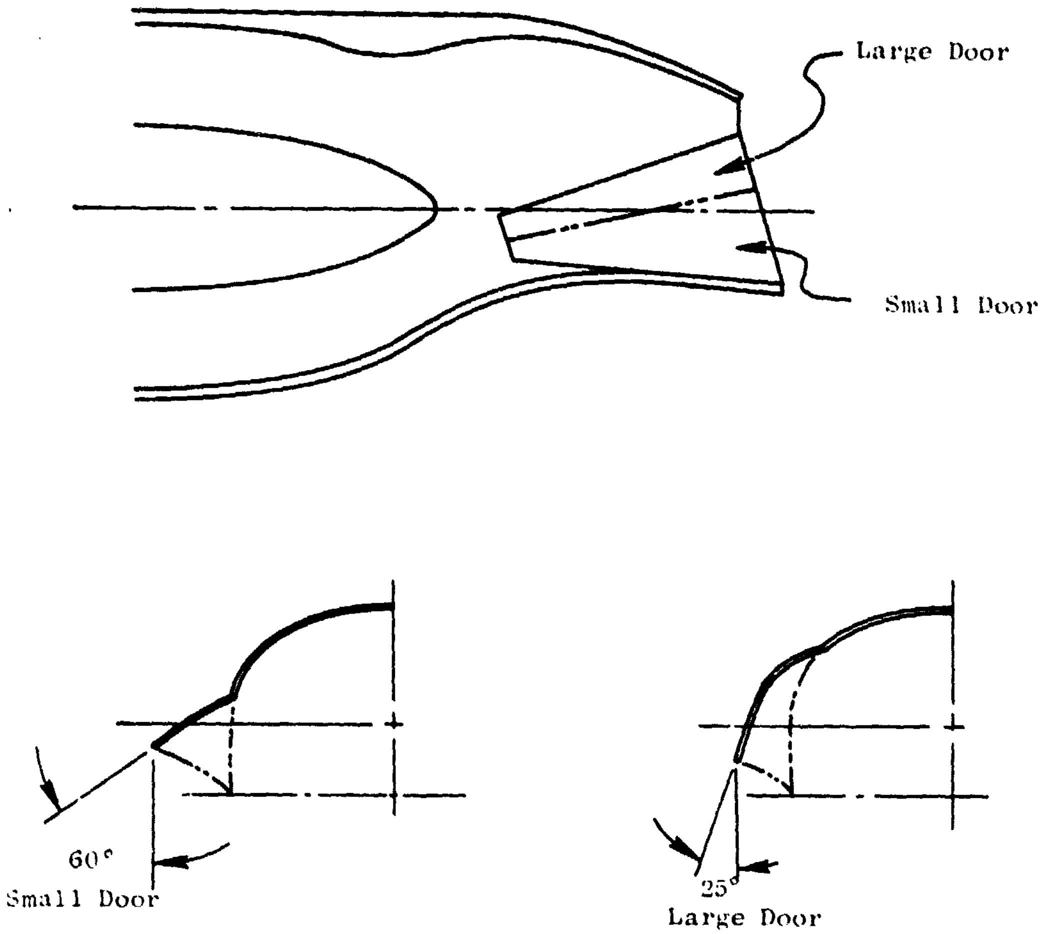


Figure 36. Baseline Door Designs.

Symbol	Nozzle	Side Doors
◇	Baseline	Large, 25°
△		Small, 60°
●	RC-1	Large, 25°
■		Small, 60°

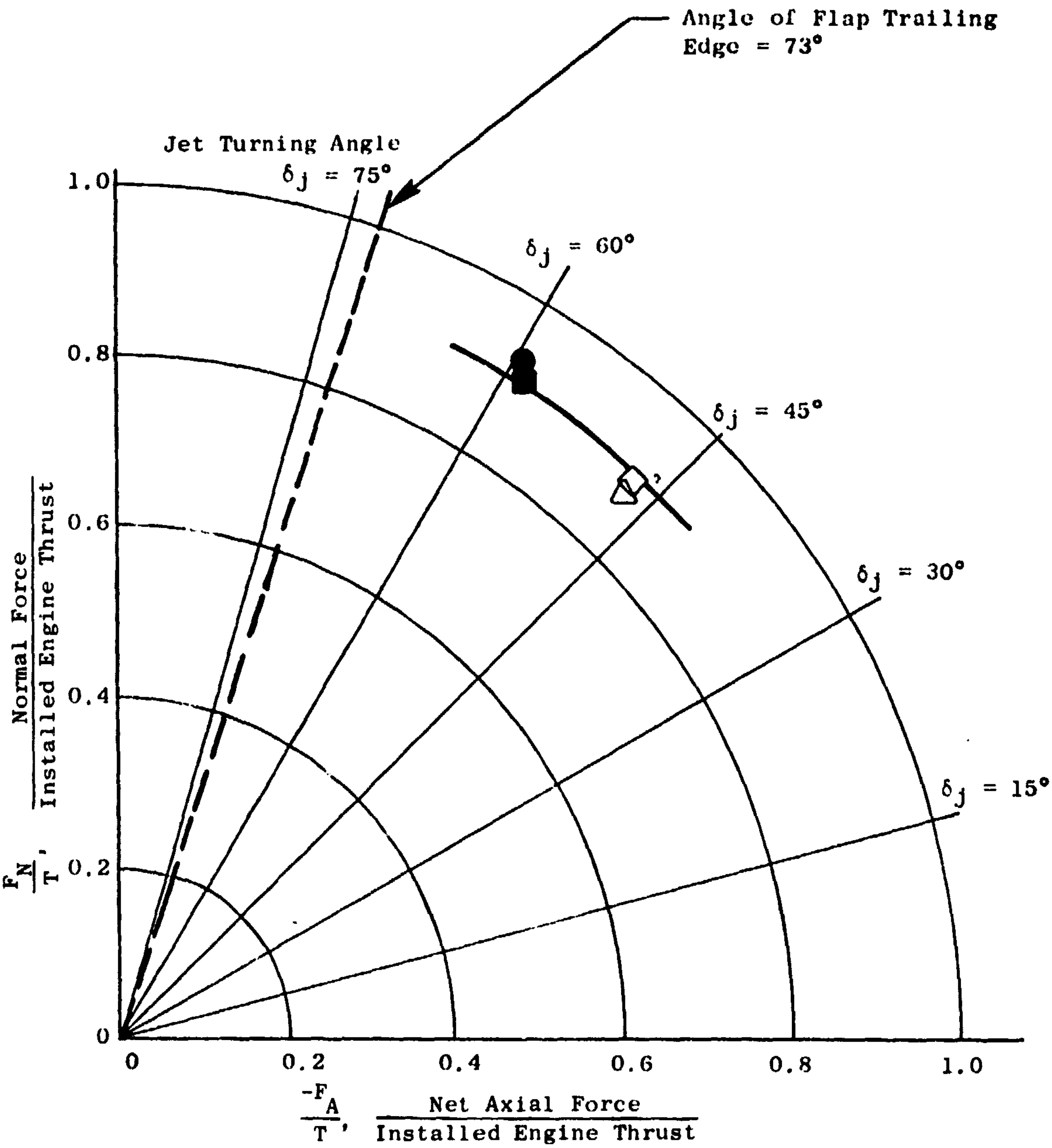


Figure 37. Baseline/Recontoured Nozzle Static Turning.

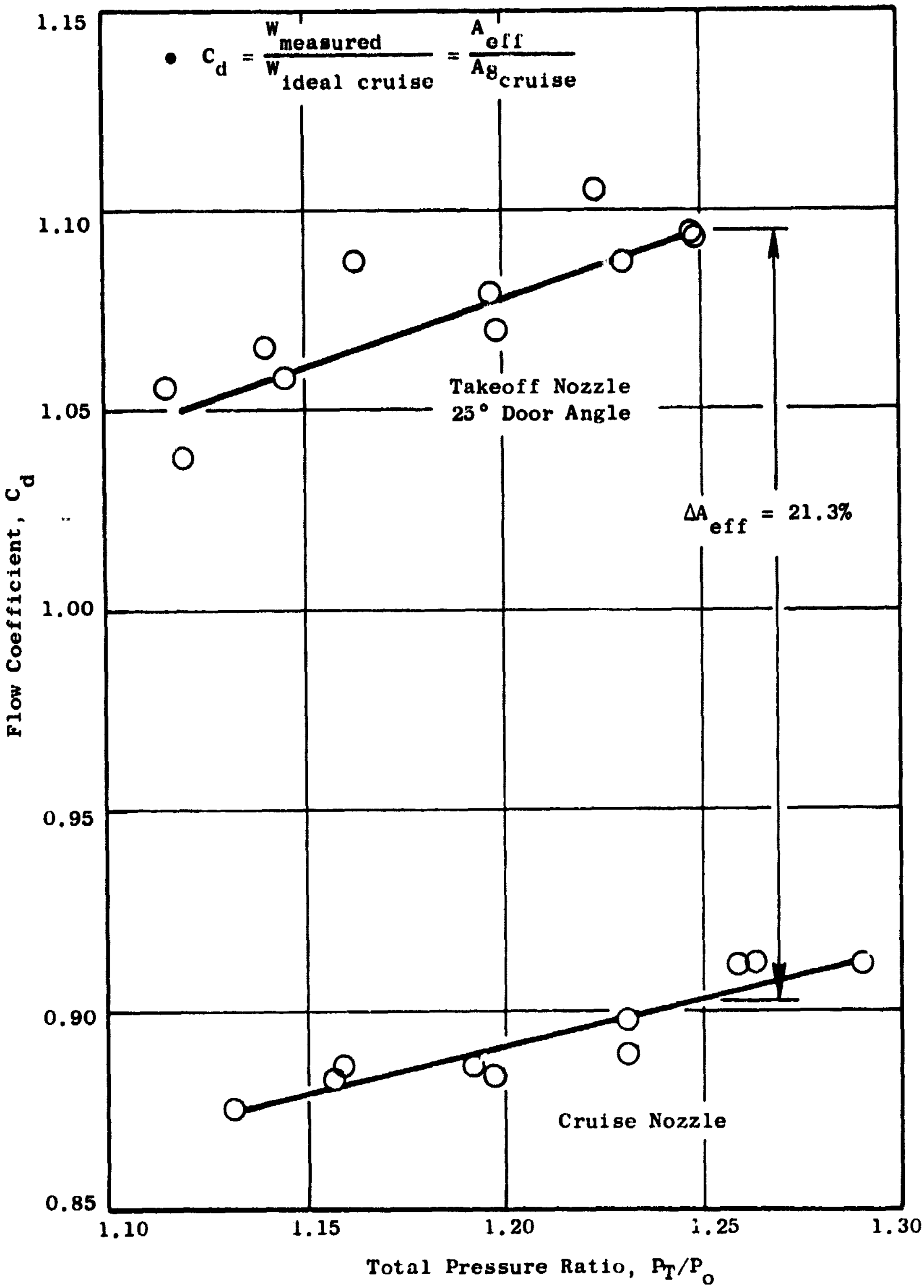


Figure 38. Recontoured No. 1 Nozzle Flow Coefficients.

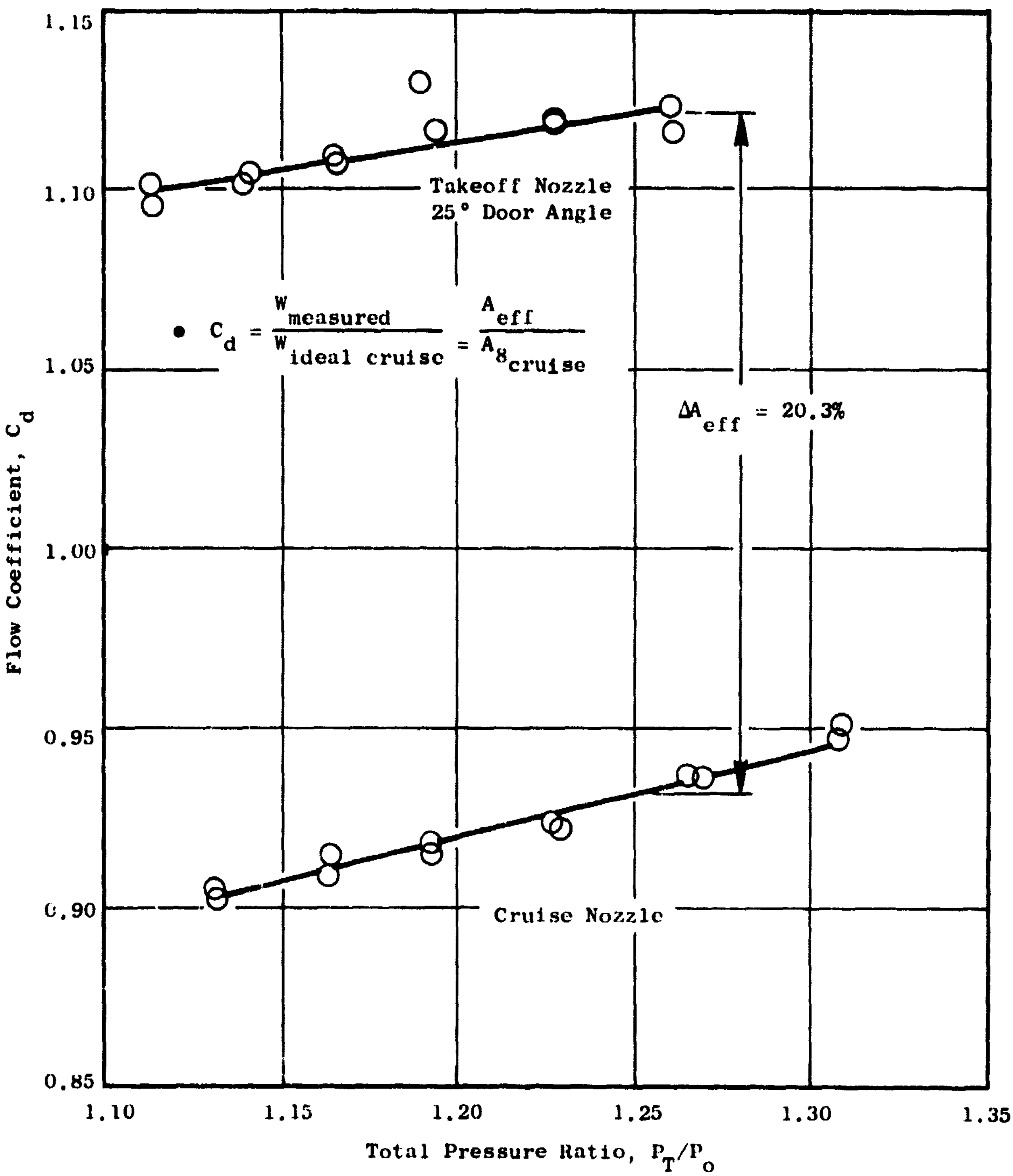


Figure 39. Baseline Nozzle Velocity Coefficients.

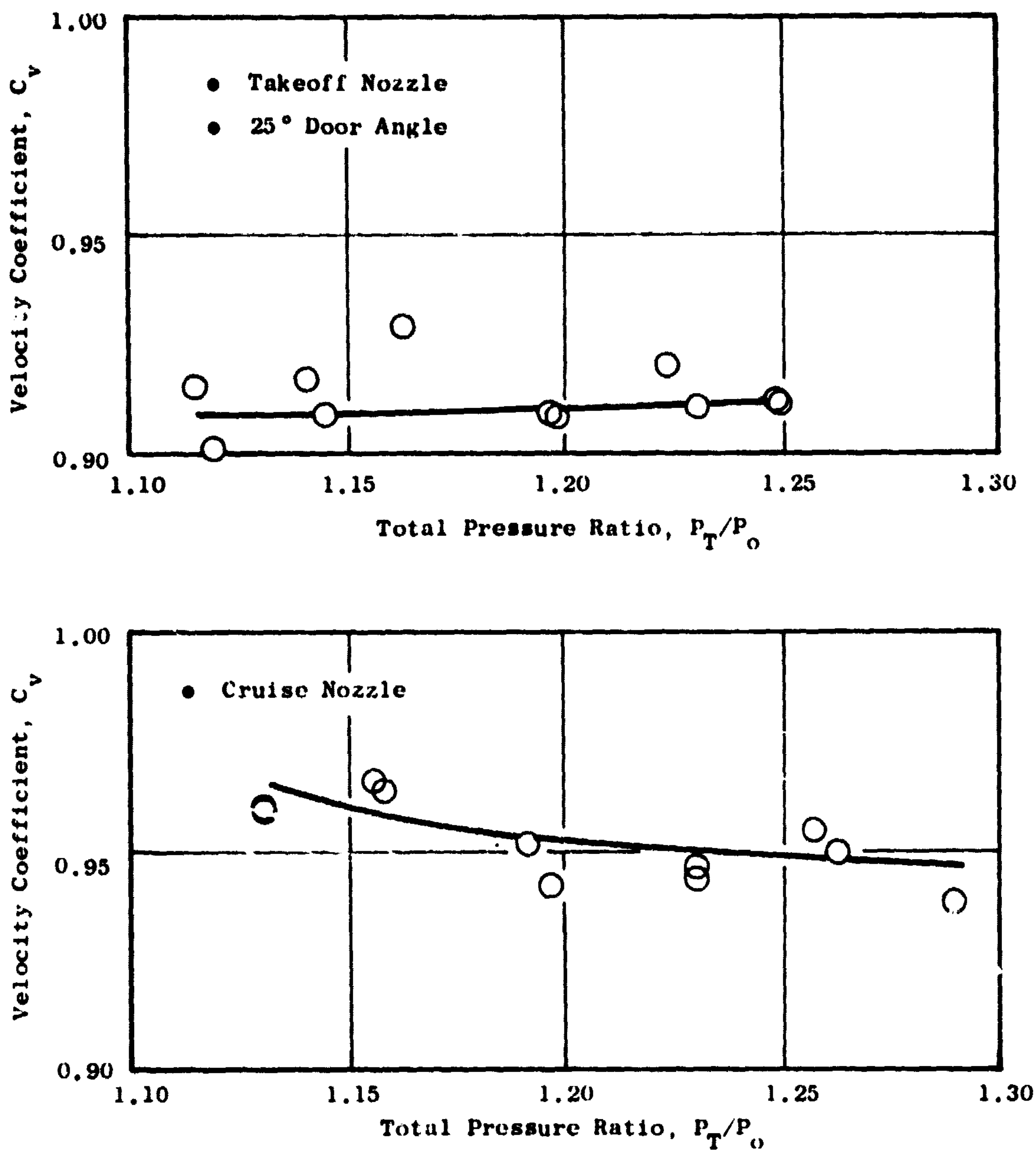


Figure 40. Recontoured Nozzle No. 1 Velocity Coefficients.

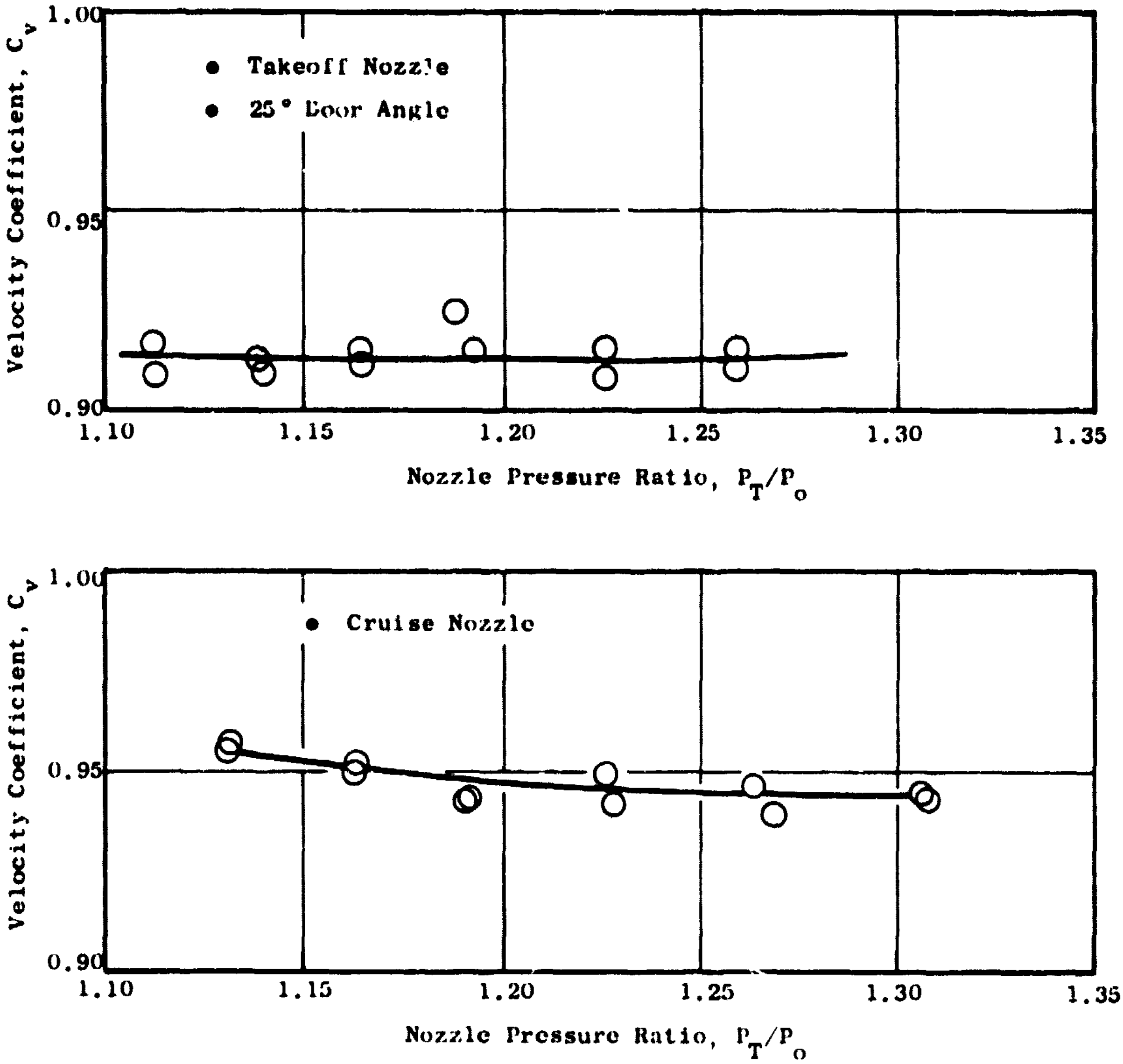


Figure 41. Baseline Nozzle Velocity Coefficients.

The wind-on evaluation was conducted in the NASA-Langley 3.65 m (12 ft) pressure tunnel. The test wing and flap arrangement is shown in Figure 42 schematically and installed in the wind tunnel in Figure 43. Illustrated in Figure 44 is the model with tufts under actual freestream conditions. The tufts show excellent flow attachment and confirm that the excellent static flow attachment characteristics are retained with wind-on.

The Langley program also evaluated the impact of the addition of vortex generators. The data shown in Figure 45 with the baseline nozzle show a slight improvement with the vortex generators but the system lift was still less than the recontoured nozzles and, therefore, a recontoured nozzle was selected as the QCSEE design. Another factor in the selection was the unknown impact of vortex generators on noise generation.

The QCSEE OTW propulsion system experimental engine final flow-path design which evolved from the Langley development tests is designated RC-1A, being derived from the roof lines from RC-1 and the floor lines of RC-2 (see Figure 36). Roof lines from RC-1 were selected because of potentially higher cruise performance with the shallower 28°30' boattail angle versus 33° for RC-2, as shown on Figure 34. The floor lines from RC-2 were selected because the curvature was more moderate and, therefore, more conducive to flow attachment inside the nozzle.

The RC-1A flow lines were also selected for use on the preliminary OTW flight propulsion system as shown in Figure 23. Although this configuration demonstrated the static turning and low speed turning performance characteristics representative of a good flight design, high speed cruise wind tunnel test results (Reference 11) obtained by NASA-Lewis showed a considerable drag penalty. These wind tunnel data indicated a need for further nozzle optimization studies (beyond the present scope of the QCSEE contract) to arrive at a better-balanced flight design which compromises the static and low speed turning performance in favor of higher cruise performance. These studies should include nozzle roof lines with shallower boattail angles and more rounded external cross-sectional contours. Use of vortex generators should again be considered to recover the reduction in turning performance which results from the recontouring. The vortex generator would retract during cruise flight. Also, depending on the selection of an aircraft configuration and nacelle placement, the nozzle side door geometry may require redefinition to provide cycle area matching with flow spreading characteristics which avoid potentially adverse aircraft/exhaust field interaction.

4.3.3 Thrust Reverser

The OTW thrust reverser configuration was established from 1/12 scale model tests at the NASA-Langley Research Center. The test setup shown schematically in Figure 46 was the same facility used for "D" nozzle development. Reverser configurations were attached to the aft portion of a single-stream simulation of the QCSEE OTW fan duct and "D" nozzle model which was modified to form the basic reverser opening and blocker door

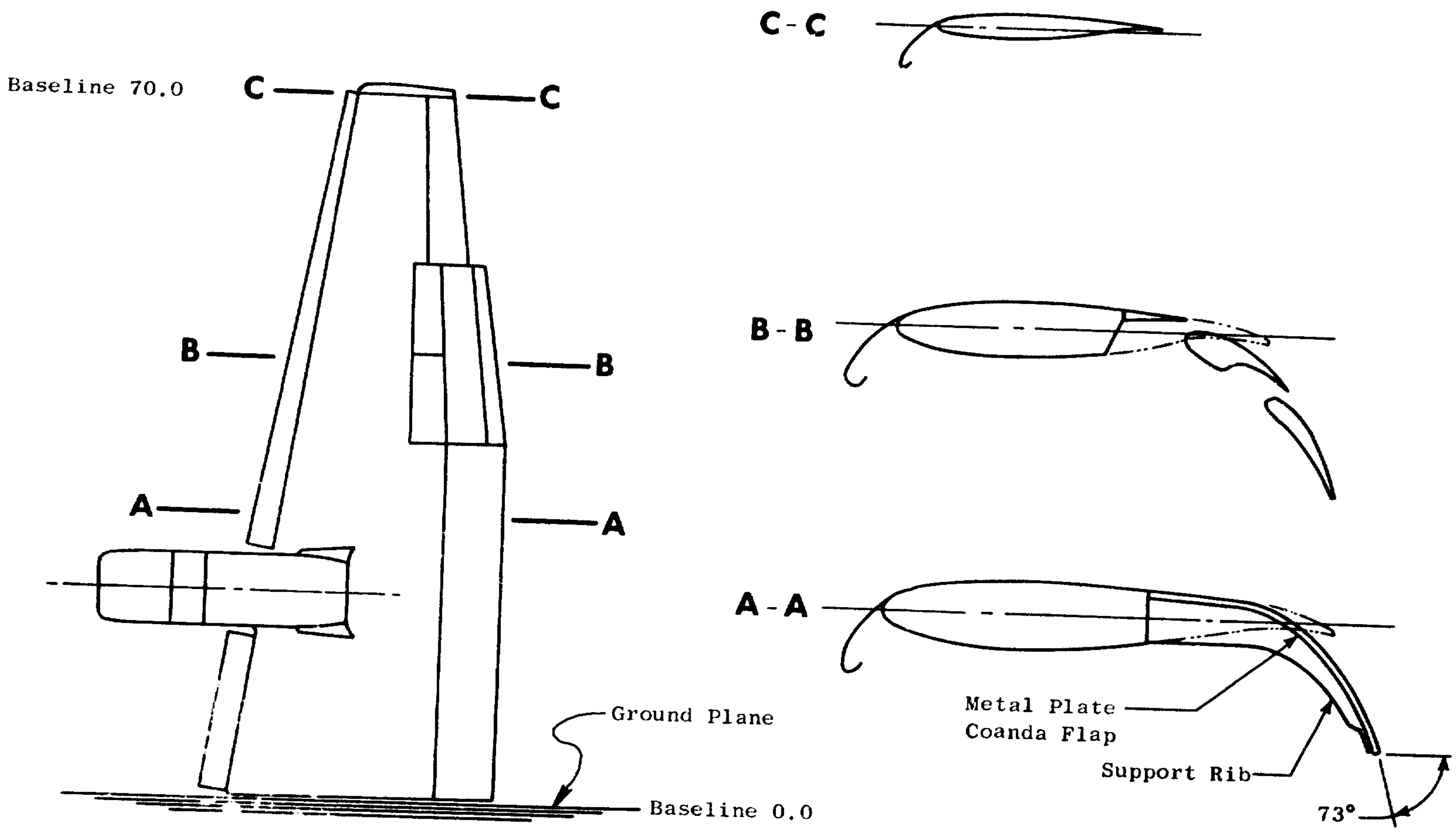


Figure 42. Model Used in Wind Tunnel Investigation (Schematic).

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Figure 43. Model Used in Wind Tunnel Investigation.

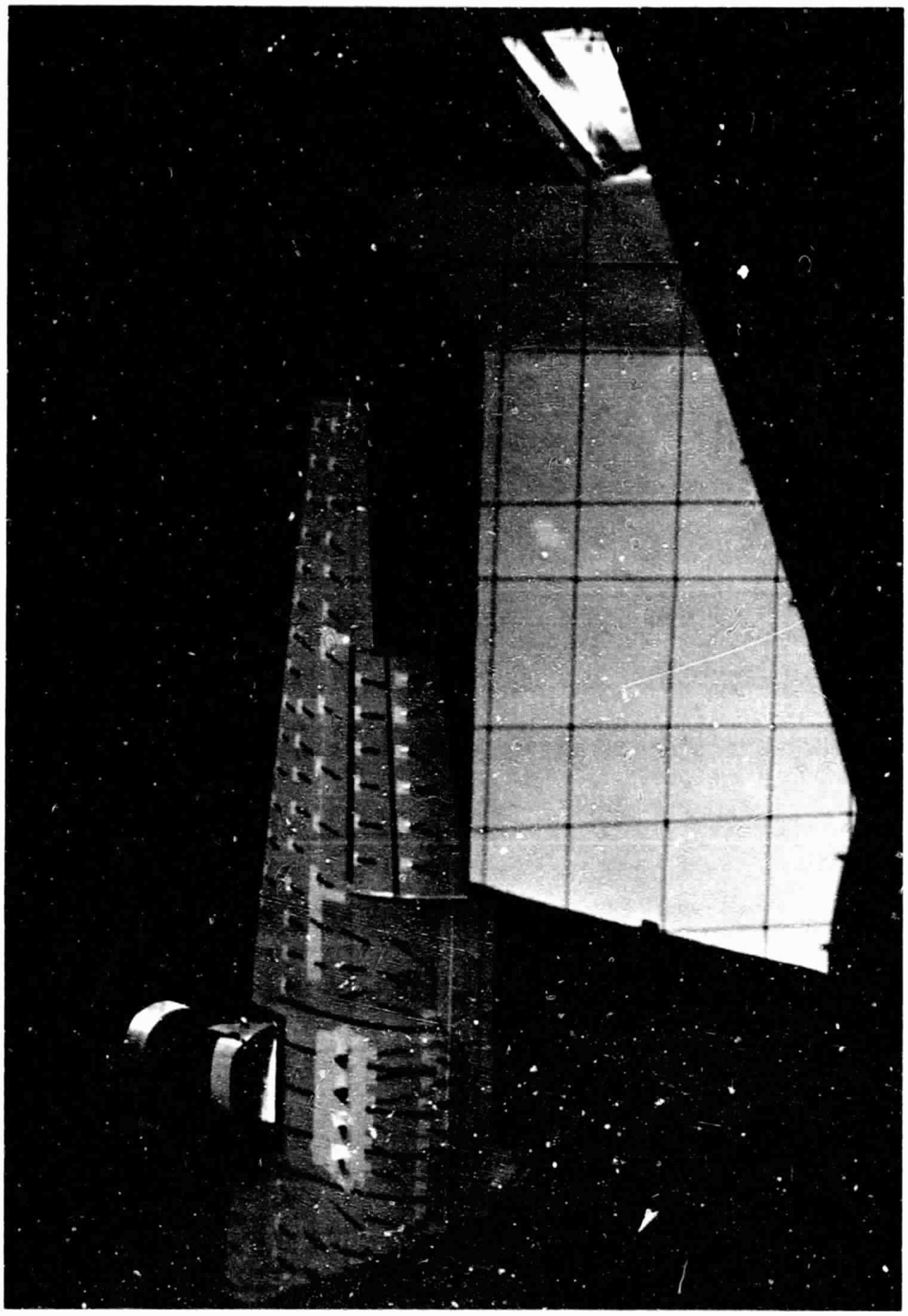

Figure 44. Jet Turning Flow Visualization.

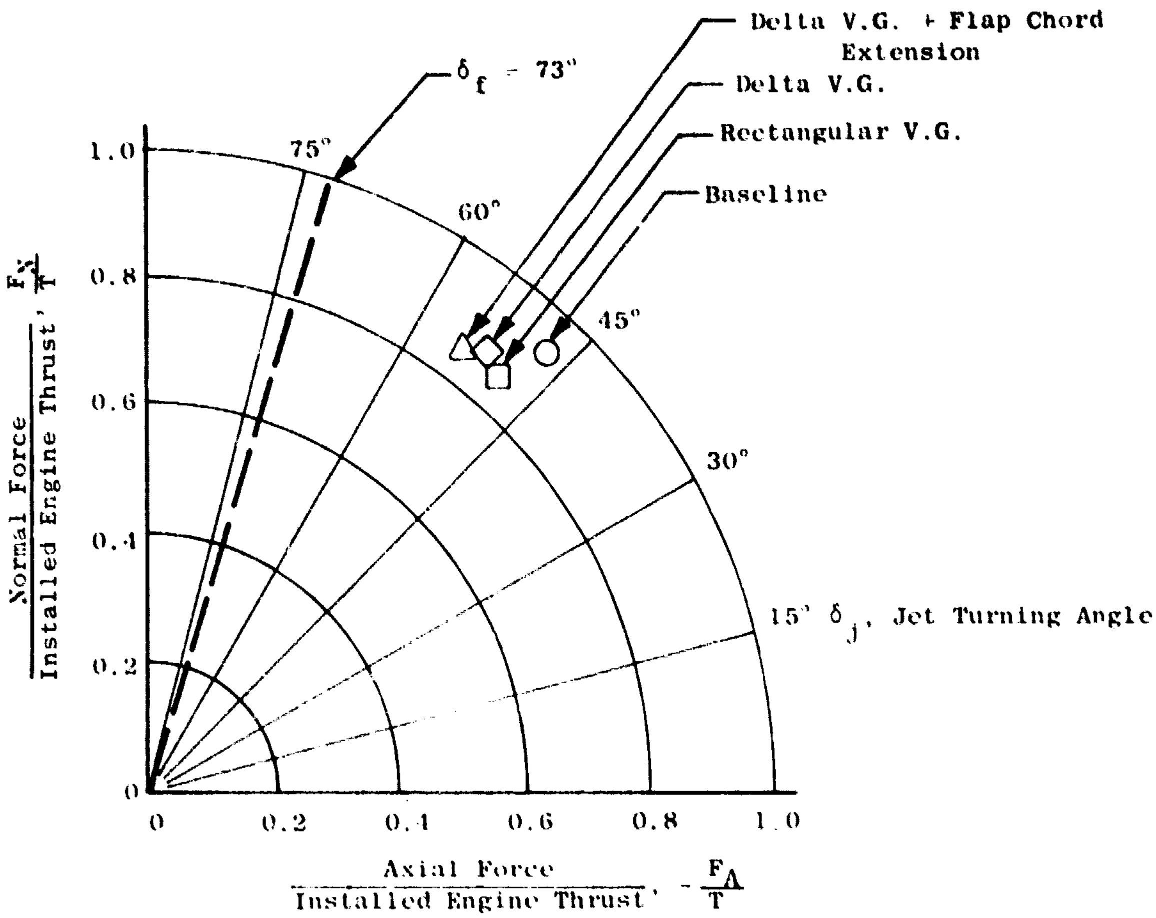

Figure 45. Effect of Vortex Generators (V.G.) on Static Turning with Baseline Nozzle.

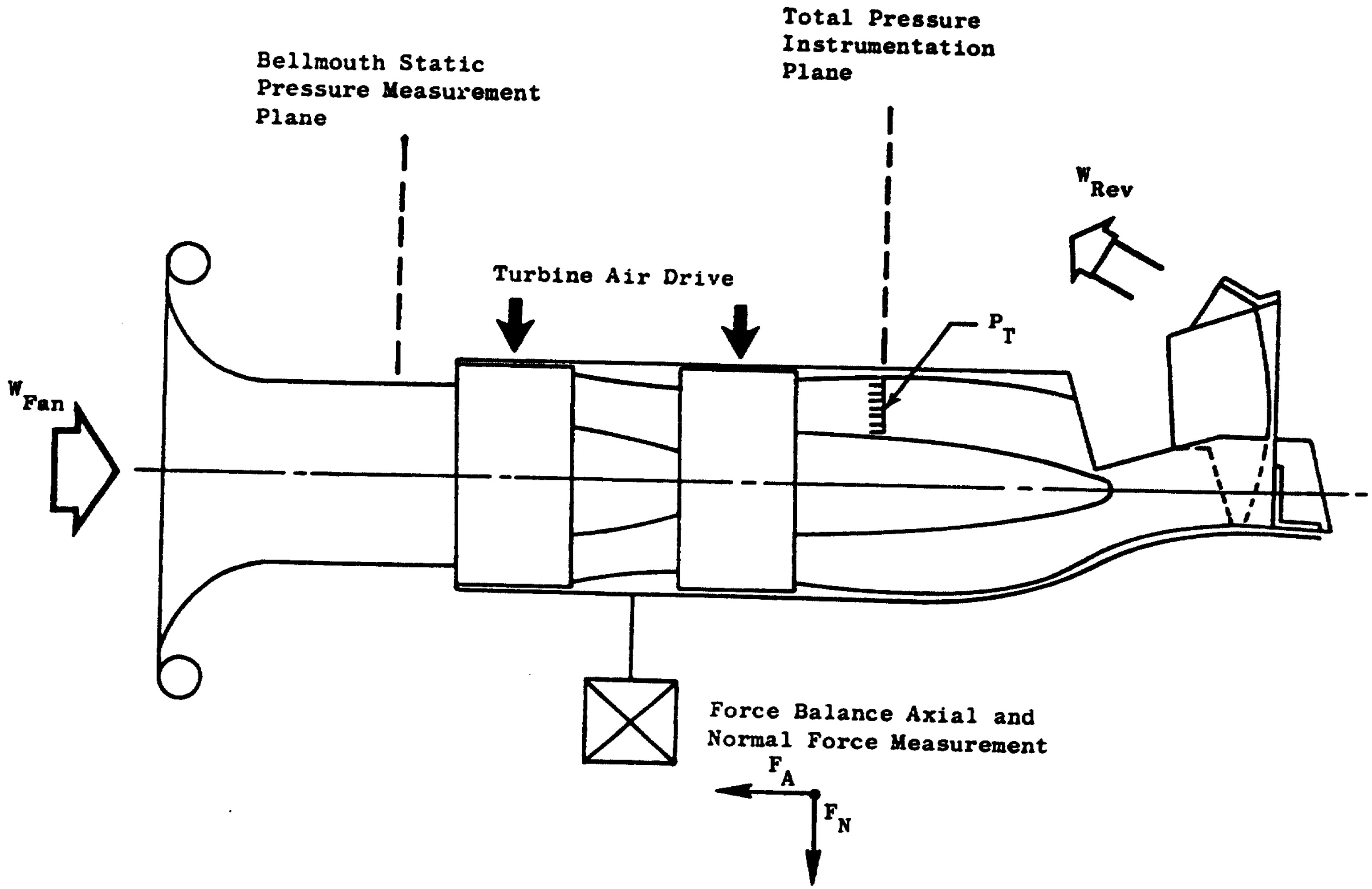

Figure 46. Reverse Thrust Static Test Configuration Schematic, Tandem Fan.

assembly. A typical scale model installation in the static facility is shown on Figure 47. This figure also shows the blast shield used to prevent reingestion.

Reverse geometry investigated are shown on Figure 48, and included blocker axial spacing (X), blocker height (H_B), lip length (L) and lip angle (β), blocker door rotation angle (α), and skirt geometry and rotation angle (ϕ). Figure 48 further describes the geometric parameters evaluated. Forty-three configurations were tested in all, with reverser pressure ratios ranging between 1.12 and 1.32. References 2 and 10 provide detailed information on model design and the experimental results obtained from this program.

The OTW reverser configuration selected from analysis of the scale model data is shown in Figure 49. The configuration has an increased lip length (L/D_{TH} = 0.4) and the lip angle 'β' was reduced to 25°. The reverser blocker door opens to full reverse through a 70° rotation angle, Ω, (representative of the α = 105° scale model setting) and tabbed side skirts rotated outward to ϕ = 45° are incorporated. The reverser side cut line is also trimmed (see Figure 48) to take advantage of the performance gain indicated from scale model tests. The blocker door pivot point remains unchanged at X_p/D_{TH} = 0.865, since axial spacing test results showed no significant gains in airflow capacity could be made by moving the blocker aft any reasonable amount. Scale model test results are shown in Figure 50 for the selected configuration. This figure shows that the objective reverse thrust performance (35% of takeoff thrust) is met at a pressure ratio slightly greater than the takeoff value of 1.29. The reverse mode airflow ratio of W_{Rev}/W_{Fwd} = 0.8 indicates an engine backpressure condition relative to the forward thrust condition which could not be eliminated with reasonable changes in reverser goemetry alone. Figure 50 also shows the beneficial effects of side skirt configuration and rotation angle, ϕ.

Scale-model performance data from the selected configuration (see Figure 50) was adjusted to account for blocker door leakage expected on the full-scale reverser. Airflow ratio and reverse thrust characteristics with leakage effective area of 770 cm^2 (119 $in.^2$) was used based on a physical leakage area of 1020 cm^2 (158 $in.^2$) and a flow coefficient of 0.75. The calculated full-scale reverser performance data from Figure 51 shows the 35% reverse thrust goal is met at a pressure ratio slightly above the takeoff value. These data also show that the QCSEE engine is backpressured 15% (W_{Rev}/W_{Fwd} = 0.85) in reverse mode relative to the forward thrust takeoff conditions.

While the reverser performance level demonstrated in model testing appears to be acceptable for the OTW flight propulsion system shown in Figure 23, the engine backpressure effects (reduced effective cycle area) indicated probably would be unacceptable for the final design of a flight nacelle. Therefore, additional nacelle optimization studies, including larger maximum nacelle diameter in particular, would be required to provide a final design having increased duct flow area at the reverser charging

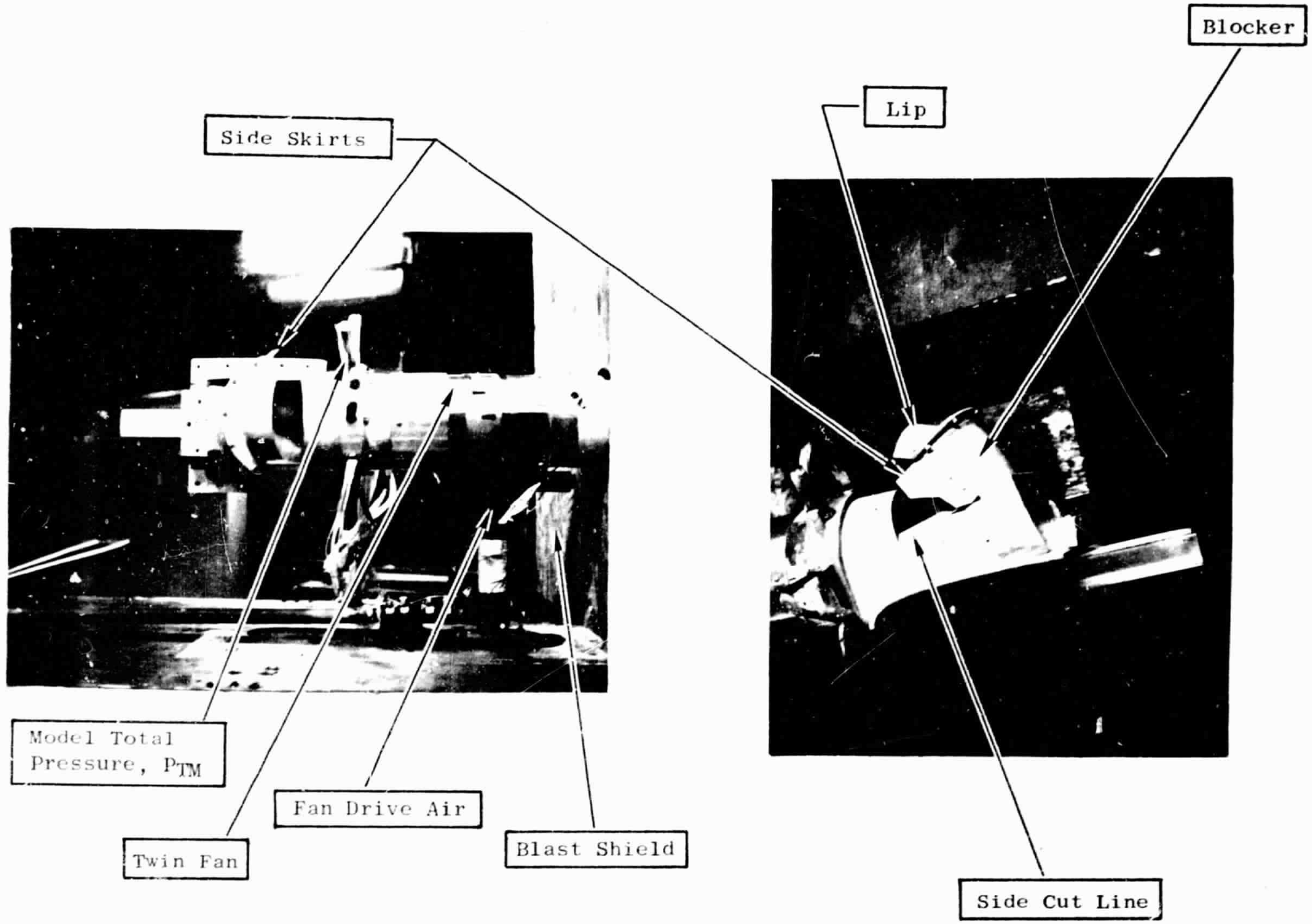

Figure 47. OTW Thrust Reverser Static Test Installation.

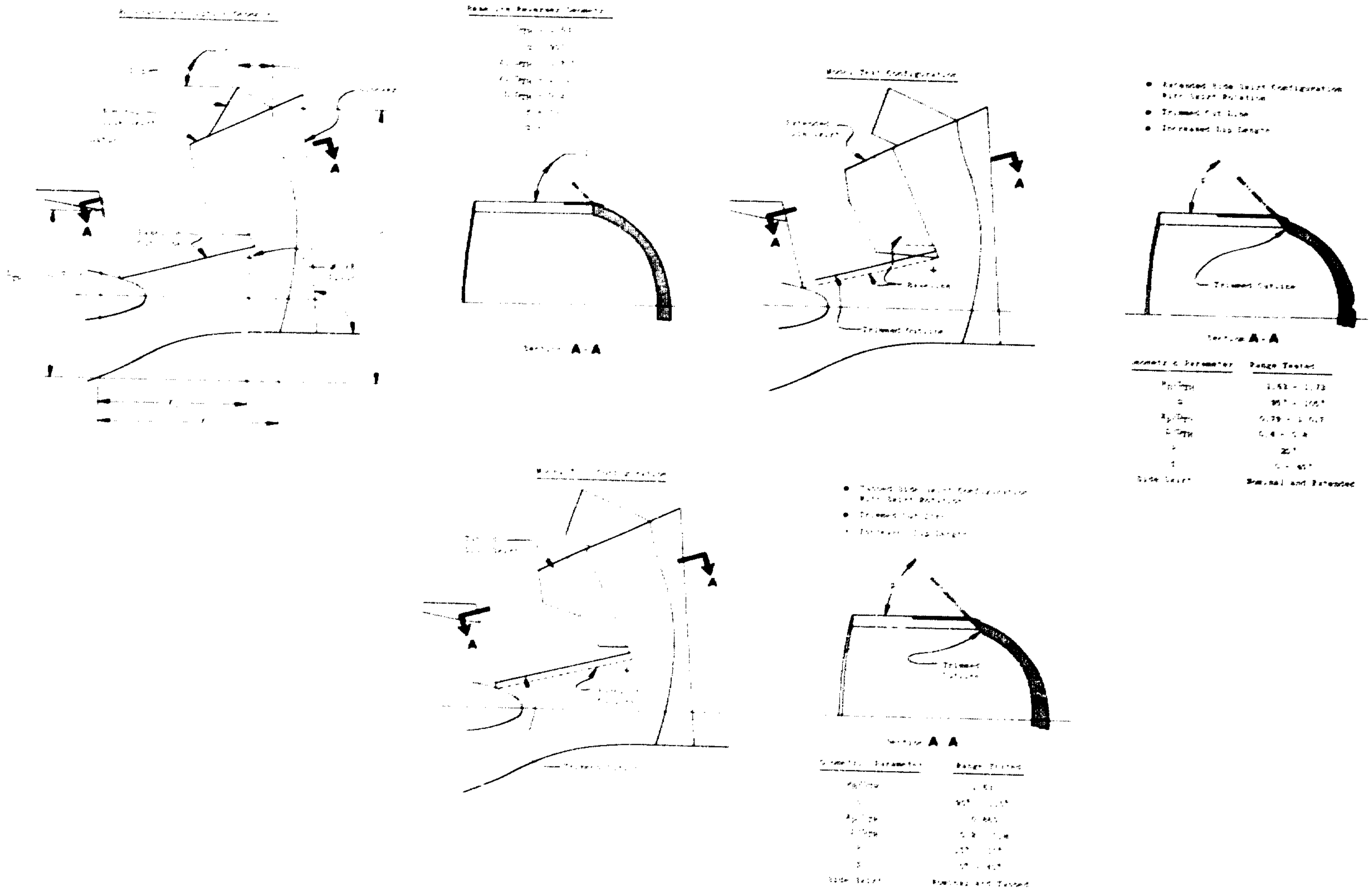

Figure 48. Thrust Reverser Scale Model Geometry.

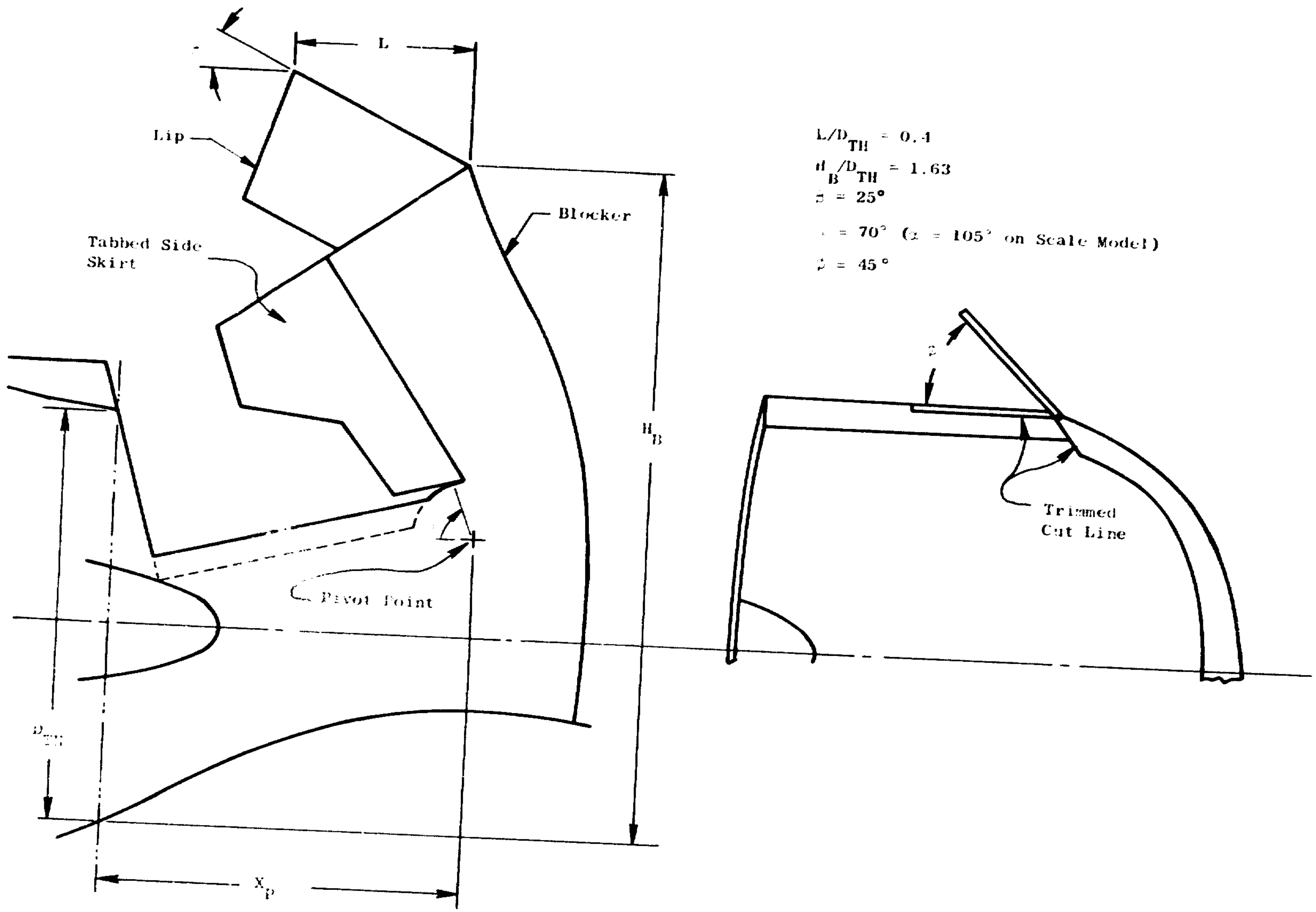

Figure 49. OTW Reverser Configuration.

Sym	Run	H_B/D_{TH}	L/D_{TH}	X_P/D_{TH}	Side Skirt	Skirt Angle ϕ	α	β	Cut Line
△	32	1.63	0.4	0.865	Nominal	45°	105°	25°	Trimmed
○	33	1.63	0.4	0.865	Tabbed	45°	105°	25°	Trimmed
□	31	1.63	0.4	0.865	Tabbed	0°	105°	25°	Trimmed

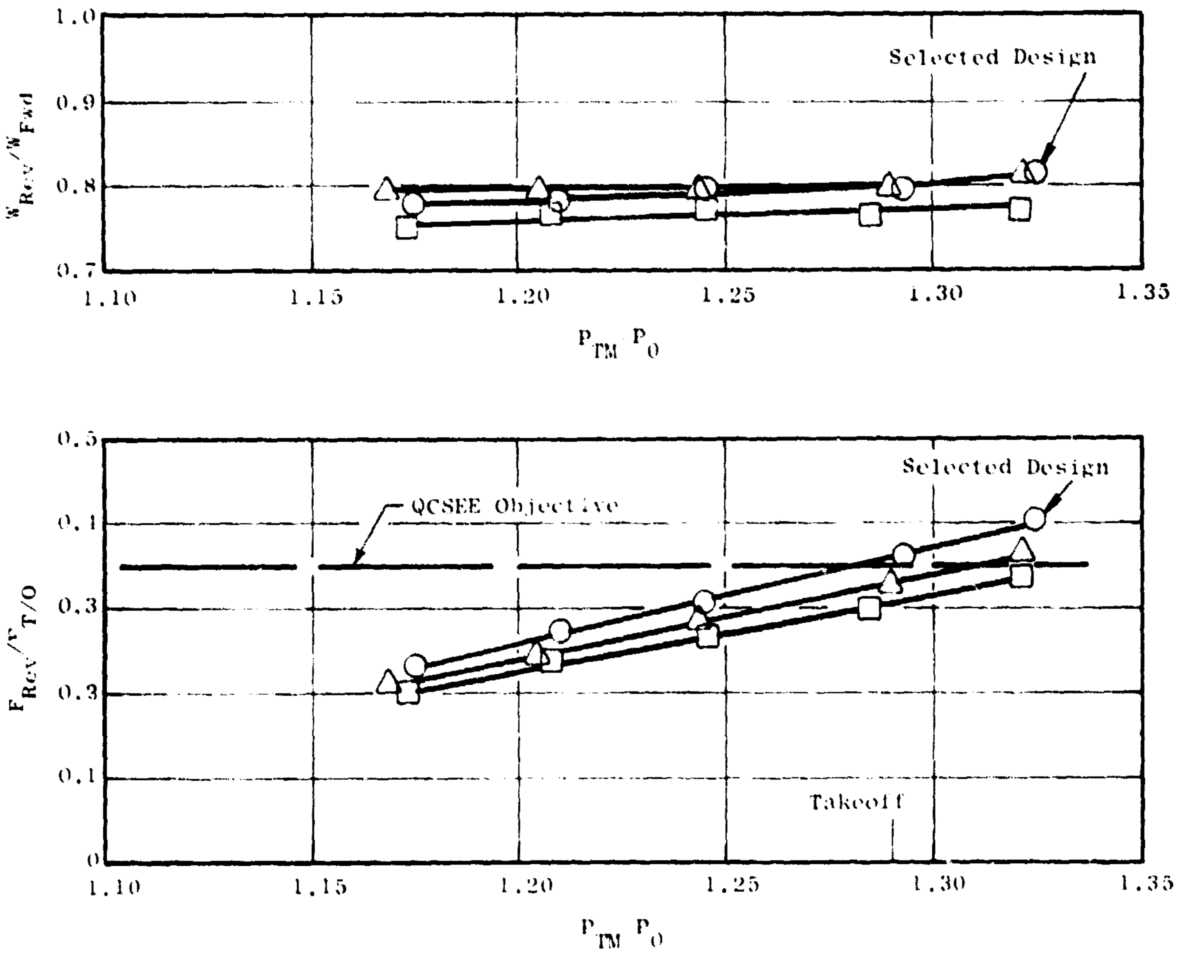

Figure 50. Scale Model Test Results of Selected Target Reverser and Various Side Skirt Geometries.

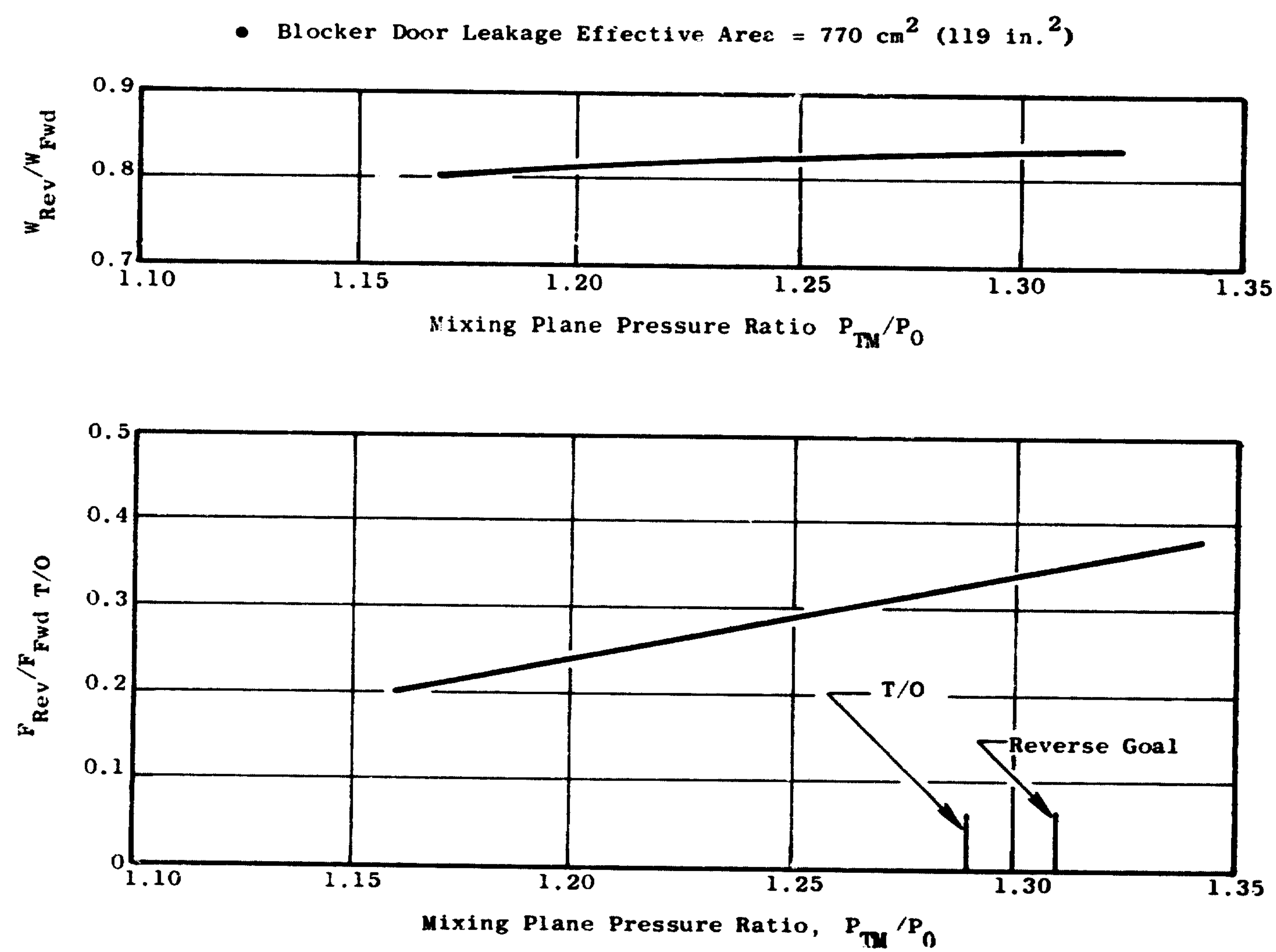

Figure 51. Estimated Reverse Thrust and Airflow Characteristics for the OTW Reverser.

station (blocker door forward trimline region) to offset the low reverser system effective area. Concurrently, the blocker door lip geometry and side skirt configuration should be reexamined to determine if lip length could be reduced and skirt rotation eliminated.

LIST OF SYMBOLS

A_{eff}	Exhaust Nozzle Effective Area
$A8_{cruise}$	Cruise Nozzle Physical Area
A_i	Inlet Throat Area
C_d	"D" Nozzle Flow Coefficient
C_v	"D" Nozzle Velocity Coefficient
D	Diffuser Exit Diameter
D_{max}	Nacelle Maximum Diameter
D_{TH}	Reverser Charging Station Height
F_A	Force Balance Axial Thrust Component
F_N	Force Balance Normal Thrust Component
F_{Rev}	Reverse Thrust
$F_{T/O}$	Forward Thrust at Takeoff (P_T/P_o = 1.29)
H_B	Reverser Blocker Door Height
L	Blocker Lip Length
L_d	Inlet Diffuser Length
M_i	Inlet Throat Mach Number
P_o	Ambient Static Pressure
P_T	Nozzle Reverser Model Total Pressure
P_{TM}	Engine Cycle Fan and Core Stream Mixing Plane Total Pressure
$P_{T_{max}}$	Maximum Inlet Total Pressure
$P_{T_{min}}$	Minimum Inlet Total Pressure
$P_{T_{avg}}$	Average Inlet Total Pressure
R	Radius
R_i	Inlet Throat Radius

C-2

LIST OF SYMBOLS (Continued)

R_{HL}	Inlet Highlight Radius
R_{max}	Inlet Cowl Maximum Radius
T	Installed Thrust, Including Wing Upper Surface Scrubbing and Spreading Effects
V_{cw}	Inlet Crosswind Velocity
V_o	Freestream Velocity
V.G.	Vortex Generators
$W_{ideal\ cruise}$	Ideal Airflow for Cruise Nozzle Physical Area
W_{Fwd}	Forward Thrust Airflow
$W_{measured}$	Measured Airflow
W_{Rev}	Reverse Thrust Airflow
x	Distance from Inlet Throat to Maximum Diffuser Angle
X	Axial Distance from Inlet Highlight to R_{Max}
P	Reverser Pivot Point Axial Spacing from Charging Station
α_i	Inlet Angle of Attack
α_{cw}	Inlet Crosswind Angle
β	Reverser Lip Angle
Γ_2	Referred Engine Airflow $W\sqrt{\Theta_2}/\delta_2$
δ_2	Inlet Relative Absolute Total Pressure Ratio
δ_f	Flap Trailing Edge Angle
δ_j	Jet Turning Angle at Flap Trailing Edge
Δ	Incremental Change
η_R	Inlet Total Pressure Recovery

LIST OF SYMBOLS (Concluded)

Θ_{eq}	Equivalent Conical Diffuser Half Angle
Θ_{max}	Maximum Diffuser Wall Angle
Θ_2	Inlet Relative Absolute Total Temperature Ratio
ϕ	Side Skirt Rotation Angle
Ω	Blocker Door Rotation Angle from Stow to Deploy

C-2

LIST OF SYMBOLS (Concluded)

Θ_{eq}	Equivalent Conical Diffuser Half Angle
Θ_{max}	Maximum Diffuser Wall Angle
Θ_2	Inlet Relative Absolute Total Temperature Ratio
ϕ	Side Skirt Rotation Angle
Ω	Blocker Door Rotation Angle from Stow to Deploy

C-2

5.0 NACELLE COMPONENTS AND SYSTEMS

In addition to the high-bypass turbofan engine, the OTW flight propulsion system incorporates a number of new installation concepts and advanced design components. These include:

- A composite material inlet providing a high (0.79) throat Mach number and integral acoustic treatment to suppress forward radiated fan and core noise.

- A "D"-shaped exhaust nozzle integrated with the wing, to provide nozzle area control and flow spreading, and a target type thrust reverser.

- A digital electronic fuel control incorporating fuel scheduling, compressor stator scheduling, engine safety limits, nozzle area scheduling, engine health monitoring, and interfacing with the aircraft on-board computer.

- Composite cowl doors, core cowl, aft nacelle structure, and accessory covers. Figure 52 shows the flowpath of the OTW propulsion system.

5.1 COMPOSITE COMPONENTS

The major portion of the OTW flight nacelle, with exception of the core cowl, operates at very modest temperatures, less than 355 K (180° F), permitting use of a wide variety of composite materials. The primary composite material selected for these areas consists of a woven Kevlar 49 fabric impregnated with an epoxy resin system. This material exhibits light weight, good tensile strength, moderate stiffness, and excellent impact strength. Its major drawback is its poor compressive strength, therefore, in areas requiring higher compressive capabilities, woven glass cloth is substituted for the Kevlar. Where this is necessary, the standard 7781 weave "E" glass is used, impregnated with the same matrix system as the Kevlar. Where acoustic face sheet porosities in excess of 10% open area are required, graphite is used to permit laser drilling.

For the core cowl, which must operate at elevated temperatures, a graphite/polyimide system is used allowing long term operation at 561 K (550° F). The honeycomb core material in the low temperature areas is corrosion resistant 5052 aluminum core. For the higher temperatures in the core cowl, HRH 327 glass/polyimide core is used. The honeycomb core in the acoustically treated panels is slotted to provide drainage. The specific nacelle components utilizing composite materials are the inlet, outer cowl doors, thrust reverser blocker, nozzle area control doors and access panels, and the inner cowl. These are discussed individually below.

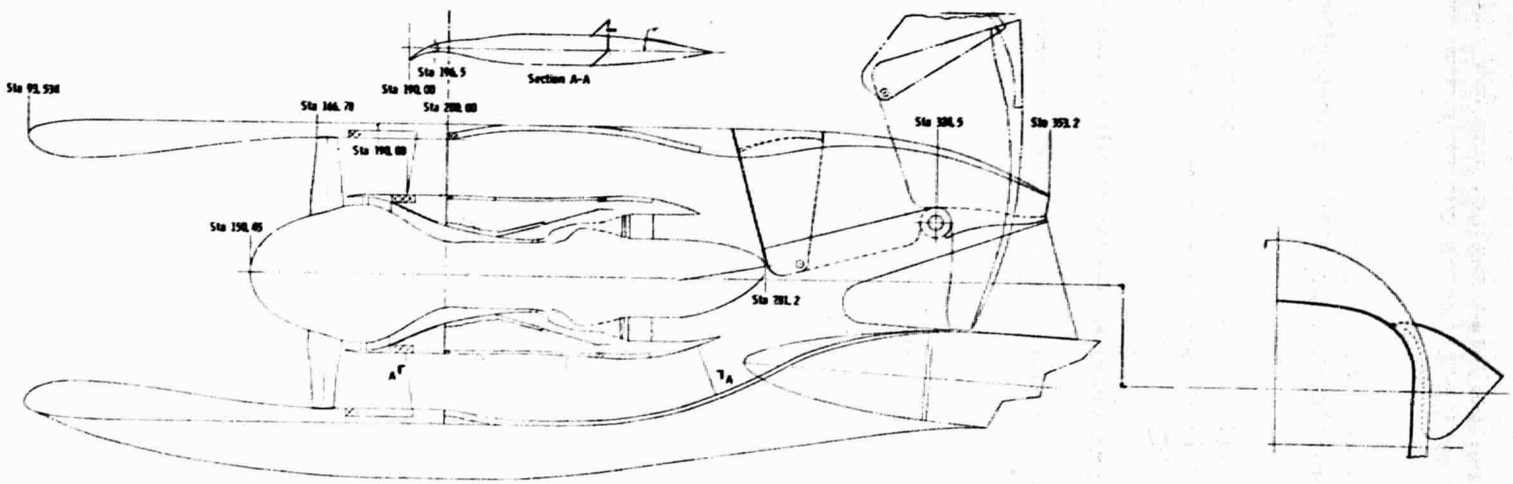

Figure 52. OTW Propulsion System Flowpath.

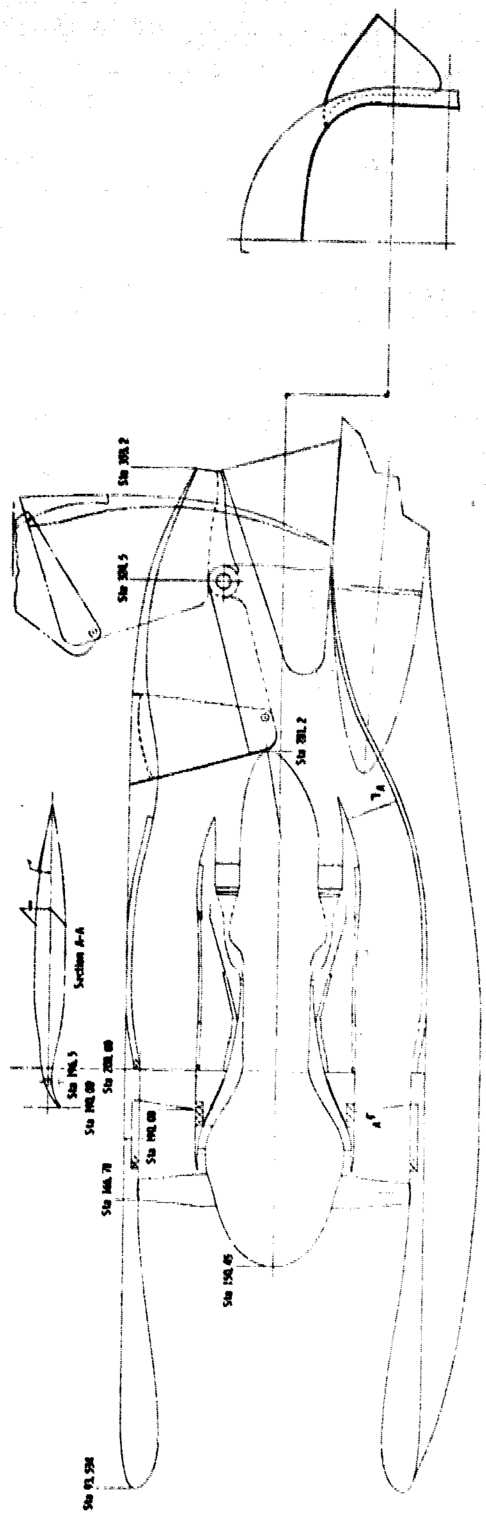

Figure 52. OTW Propulsion System Flowpath.

5.1.1 Inlet

The OTW QCSEE inlet is the largest single piece of the overall nacelle structure, being almost 178 cm (70 in.) long and nearly 200 cm (79 in.) in diameter, and has extensive acoustic treatment. To reduce the weight of this large structure as much as possible, it will be constructed mainly of lightweight Kevlar/epoxy material and the acoustic treatment will be incorporated as part of the permanent structure.

The inlet consists primarily of inner and outer honeycomb sandwich walls separated and supported by circumferential stiffeners as shown in Figure 53 The face sheets of these sandwiches are all made from Kevlar/epoxy. The inner skin of the inner wall is perforated with hole patterns to provide 10% open area to satisfy acoustical requirements. The inner wall thickness (honeycomb depth) is also tailored to acoustical requirements. The outer wall thickness is sized to provide adequate stiffness. Honeycomb material is aluminum.

Aerodynamic loading of the inlet is far more significant than inertia loading. The primary cause for this is the large transverse load reaction on the inlet as it turns the entering engine airflow during any flight condition in which the direction of the free stream air is not parallel to the inlet axis. In contrast, the lightweight structure of the inlet produces relatively low inertia loads. The most severe aerodynamic loads occur during a 3 g stall, sea level, at a flight Mach number of 0.4, and maximum continuous engine power, as shown in Table VII. For design analysis, the loads resulting from this condition were combined with the most severe additive inertia loads caused by dynamic landing. In addition, compressive hoop loads were considered for the sea level static takeoff power operating condition. The stress levels for these loads and this construction are shown in Table VIII. These are based on each facing consisting of three plies of woven Kevlar/epoxy material giving a total face sheet thickness of 0.084 cm (0.033 in.). Buckling allowables for this construction are shown in Table IX. The sensitivity of this configuration to local loads is shown in Figure 54. Stiffeners are segmented and are constructed of aluminum sheet with flanged weight reduction cutouts and with composite (Kevlar) flanges to provide bonded attachment to the walls. The stiffener flanges were designed to prevent the bearing load between the stiffener flange and sandwich wall from exceeding 2.48×10^6 N/m^2 (360 psi). This resulted in a flange width of 1.55 cm (0.61 inches). Using a 137.9×10^6 N/m^2 (20,000 psi) flange bending stress allowable results in a flange thickness of 0.1778 cm (0.07 inches).

The leading edge of the flight inlet is all titanium for resistance to foreign object damage and erosion, and for anti-icing provisions. This section is removable (by unbolting) from the main body. A corrugated backup sheet provides passages for anti-icing airflow. This arrangement has the advantages of isolating the anti-icing air from the composite materials and of containing the flow for effective heat transfer and minimum air usage. A sketch of this concept is shown in Figure 55. At the rear, the inlet is attached to the forward end of fan frame by means of 16 rotary latches.

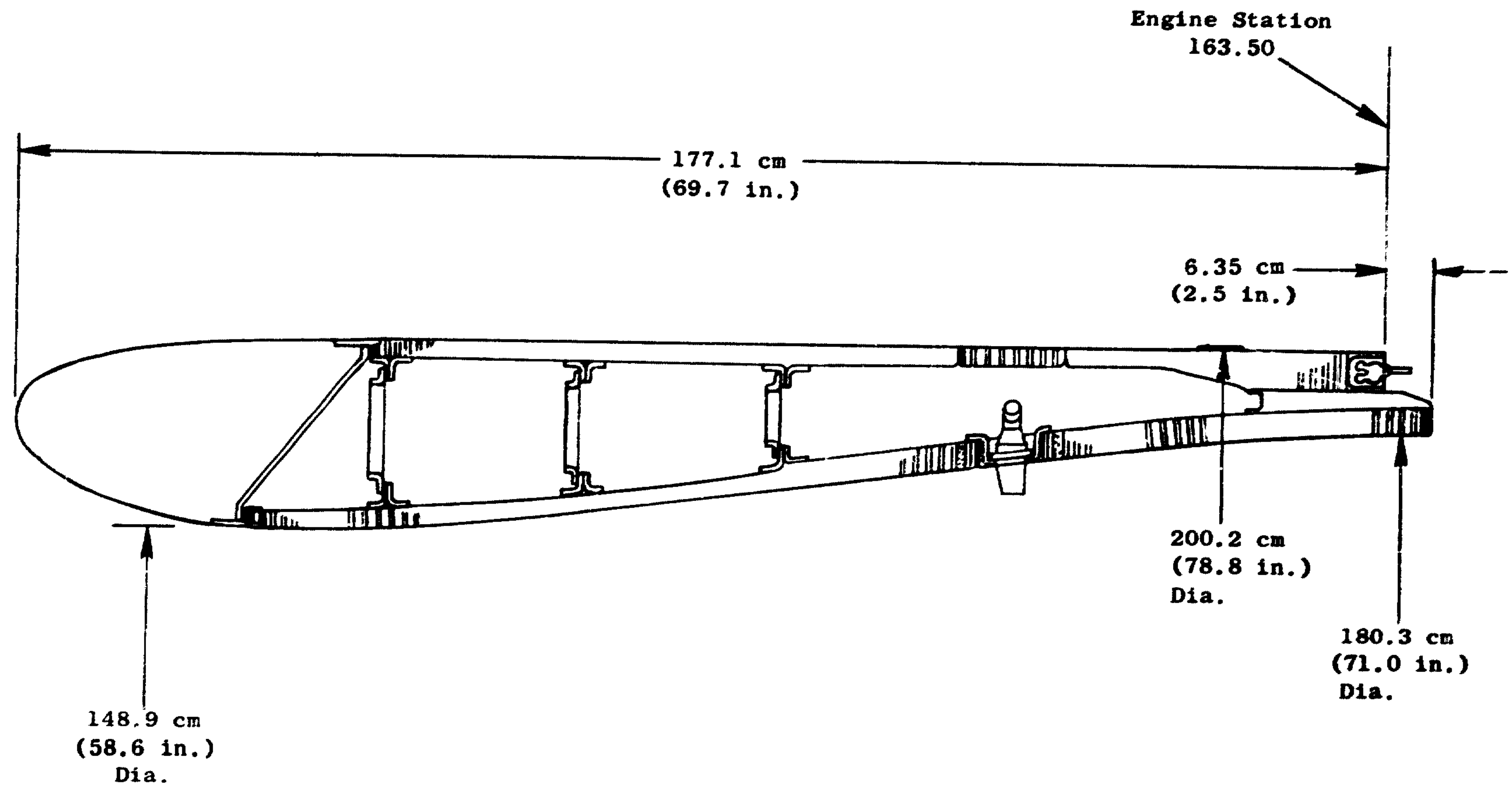

Figure 53. Inlet Axial Cross Section.

Table VII. Inlet Design Loads.

- Derived from DAC "Design - TC" Criteria Aerodynamic Loads from 3 g Stall Inertia Loads from Dynamic Stall
- Predominating Load Source:

  Aerodynamic Loads from 3 g Stall

  (98% of Latch Loads)
- Maximum Combined Loads

Bending	6,931,680 cm N (613,423 in.-lb)
Axial	8,358 N (1,879 lb)
Transverse	37,119 N (8,345 lb)

Table VIII. Inlet Stress and Deflections.*

Type	Load Magnitude	Direction	Stress			F/S	Deflection
			Type	Calculated	Allowable		
Bending	6,931,680 cm N (613,423 in.lb.)	-	Compression	1402 N/cm^2 (2034 psi)	6895 N/cm^2 (10,000 psi)	4.9	0.058 cm (0.023 in.)
Axial	8,358 N (1,879 lb.)	Forward	Tension	1583 N/cm^2 (2296 psi)	48265 N/cm^2 (70,000 psi)	30.5	
Transverse	37,119 N (8,345 lb.)	-	Shear	403 N/cm^2 (584 psi)	6895 N/cm^2 (10,000 psi)	17.1	0.206 cm (0.081 in.)
Hoop (Max Range Expected)	2.8 N/cm^2 (4 psi)	Burst	Tension	1834 N/cm^2 (2660 psi)	48265 N/cm^2 (70,000 psi)	26.3	-
	5.9 N/cm^2 (8.5 psi)	Crush	Compression	3897 N/cm^2 (5652 psi)	6895 N/cm^2 (10,000 psi)	1.8	-

*Aerodynamic loads from 3G stall (M_p = 0.4 at SL) combined with inertia loads from dynamic landing.

Table IX. Inlet Buckling Loads.

Buckling from Bending					
Load Conditions	Critical Moment		Actual Max. Moment		F/S
	cm N	in-lb	cm N	in-lb	
Moment on Outer Skins	455,667,000	40,329,928	6,930,700	613,423	65.7
Buckling from Compressive Hoop Load					
Load Conditions	Critical Pressure		Actual Pressure		F/S
	N/cm^2	psi	N/cm^2	psi	
Pressure Supported by Single Sandwich Wall (2 skins) Between Stiffeners 33.0 cm (13 in.) Apart	153.8	223	5.9	8.5	26.2
Pressure Supported by Overall Wall Structure	275.1	399	5.9	8.5	46.9
Pressure Supported by Single Outer Aluminum Nose Wall 40.6 cm (16 in.) Long with Stiffening Corrugations	351.0	509	5.9	8.5	59.9
Pressure Supported by Single Inner Aluminum Nose Wall Aft of Corrugation with Corrugation Sheet Extended to Form Doubler: 6.4 cm (2.5 in.) Long x 0.183 cm (0.072 in.) O/A Thk	16.2	23.5	5.9	8.5	2.76

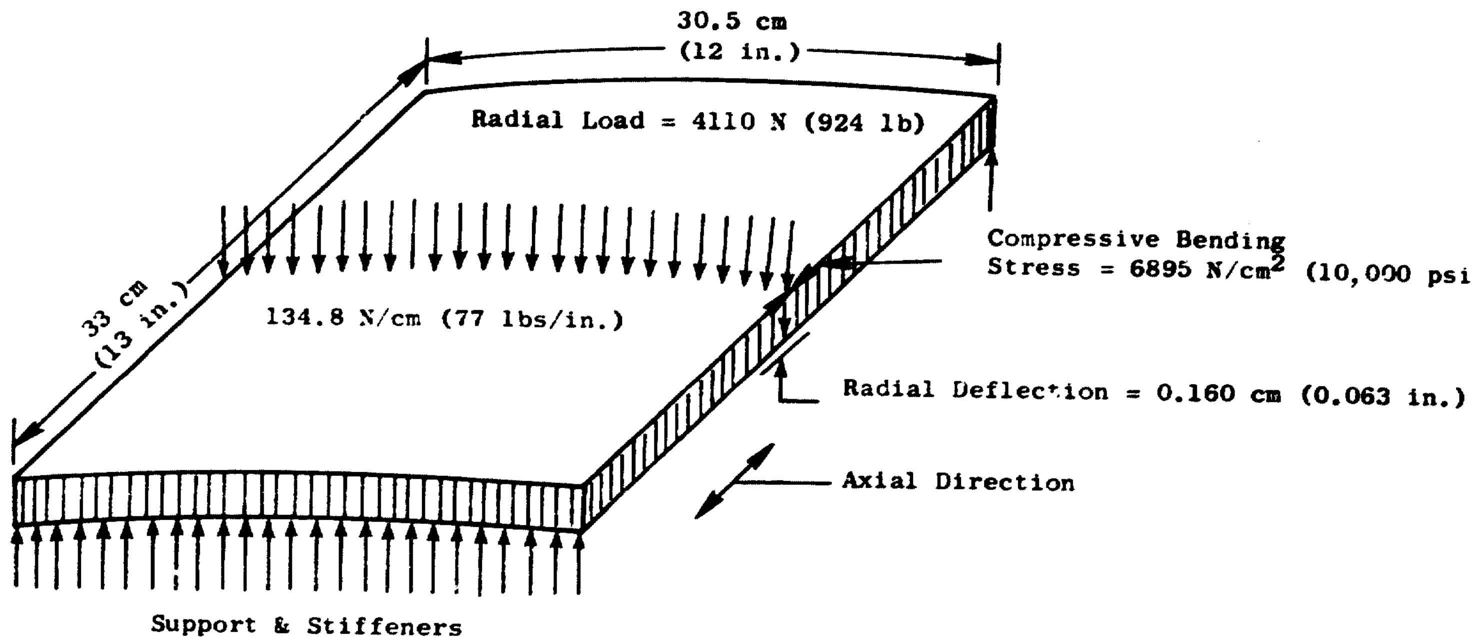

Figure 54. Inlet Wall Local Load Resistance.

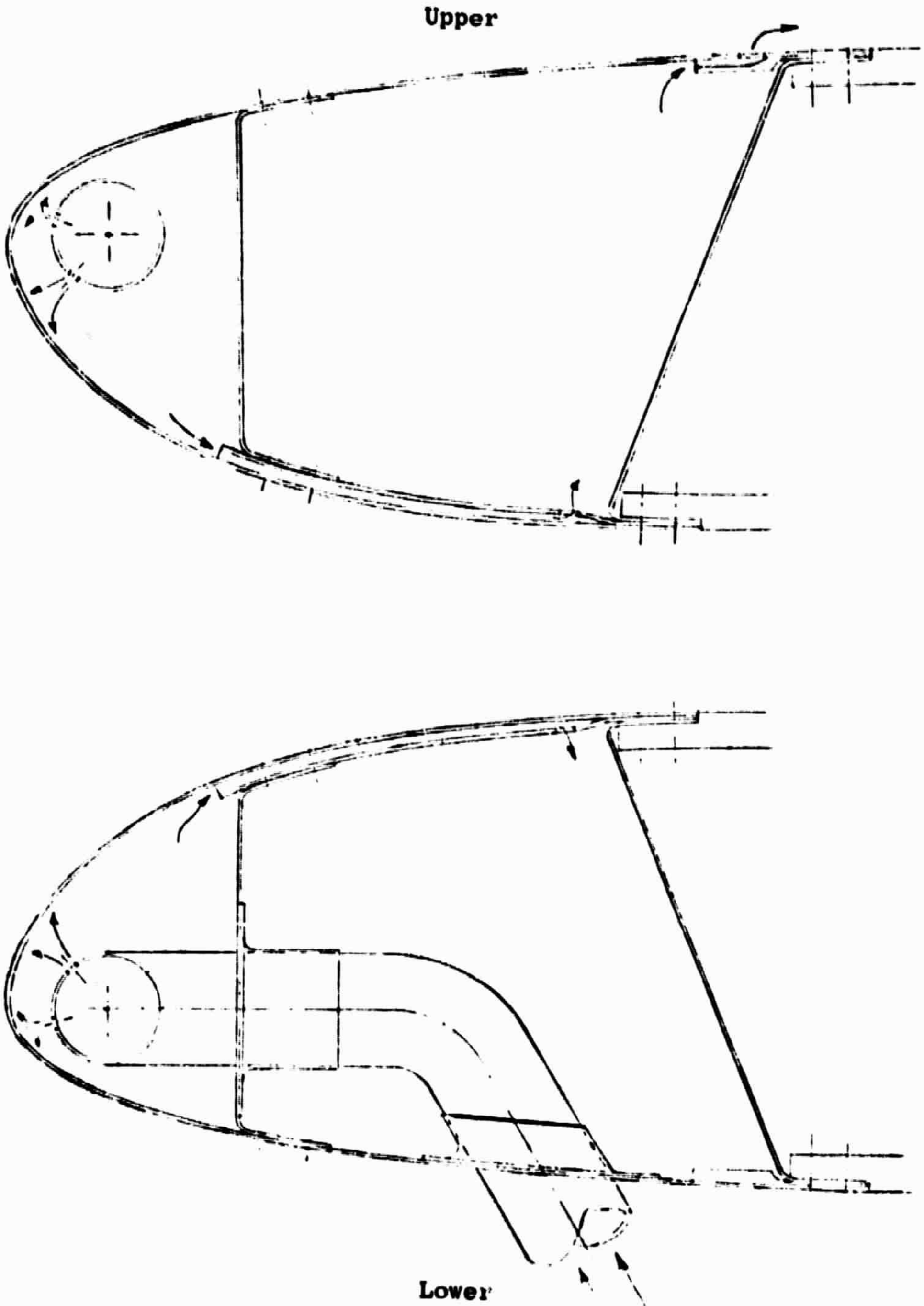

Figure 55. Inlet Lip Anti-Icing.

Each of these latches is operated by turning a flush receptacle. A pressure and acoustical seal is achieved at this joint by means of a thick (in the radial direction) elastomer gasket. The latch loads for this installation are shown in Table X.

Because of the modest stresses in the inlet, skin thickness is determined by minimum gages required for local impact loads during handling.

The composite QCSEE inlet described above weighs 156 kg (345 lb) compared to the weight of a typical current technology metal inlet (scaled to the same size) of 217 kg (479 lb).

5.1.2 Fan Bypass Duct (Fan Cowl)

The fan bypass duct (see Figure 56) is formed of the inner surfaces of the engine mount, access panels, and outer cowl doors, all of which are discussed in detail in the following sections. These components are designed to take full advantage of advanced composite materials in order to provide a lightweight, thin profile nacelle suitable for advanced air transports. In addition, all of these components are designed to allow full access to the engine either for maintenance or engine removal/installation.

Acoustic treatment is included in all areas where practiale, i.e., panels and doors. This treatment is integral with the components in order to provide a lightweight design with a maximum of treated area.

5.1.2.1 Engine Mount System

The engine mount is an over-the-engine, cage-type aft nacelle structure cantilevered forward from the aircraft wing. The loads resulting from this arrangement are shown in Table XI. The mount structure is constructed of graphite/epoxy laminations laid up to form an integrated bonded structure of hollow-cell box beams and partial rings, the skin and wall thicknesses being tailored throughout to provide the necessary section properties with a minimum of material and weight (see Figure 57). The depth of the mount structure is that available between the fan duct flow path and the nacelle outer surface. The inner and outer skins of the mount form a continuation of these flow surfaces. The two main load-carrying beam sections run the full length of the mount; these sections are joined together by overhead partial-ring sections located in the areas of the engine forward and aft mounting points. These two ring sections are further stabilized by being joined together by a lightweight graphite skin/polyimide honeycomb beam section at the top centerline. Located at the top of the forward ring, in the plane of the engine side mounts, is an integral steel lug which mates with a groove in the fan frame. This lug transfers the engine side loads to the mount structure. The engine aft mount point is attached to the top of the aft ring by means of a link which stabilizes the engine by resisting vertical loads (pitching moment) only. The engine thrust mount attachments are on the sides of the engine through

Table X. Inlet Latch Loads.

Latch Configuration	Maximum Latch Load		Ultimate Latch Strength		Latch F/S
	N	lb	N	lb	
All 16 Latched	9,617	2162	28,801	6475	3.00
One Latch Open	10,737	2414	28,801	6475	2.68
Two Latches Open	13,135	2953	28,801	6475	2.19
Six Latches Open	25,073	5637	28,801	6475	1.15

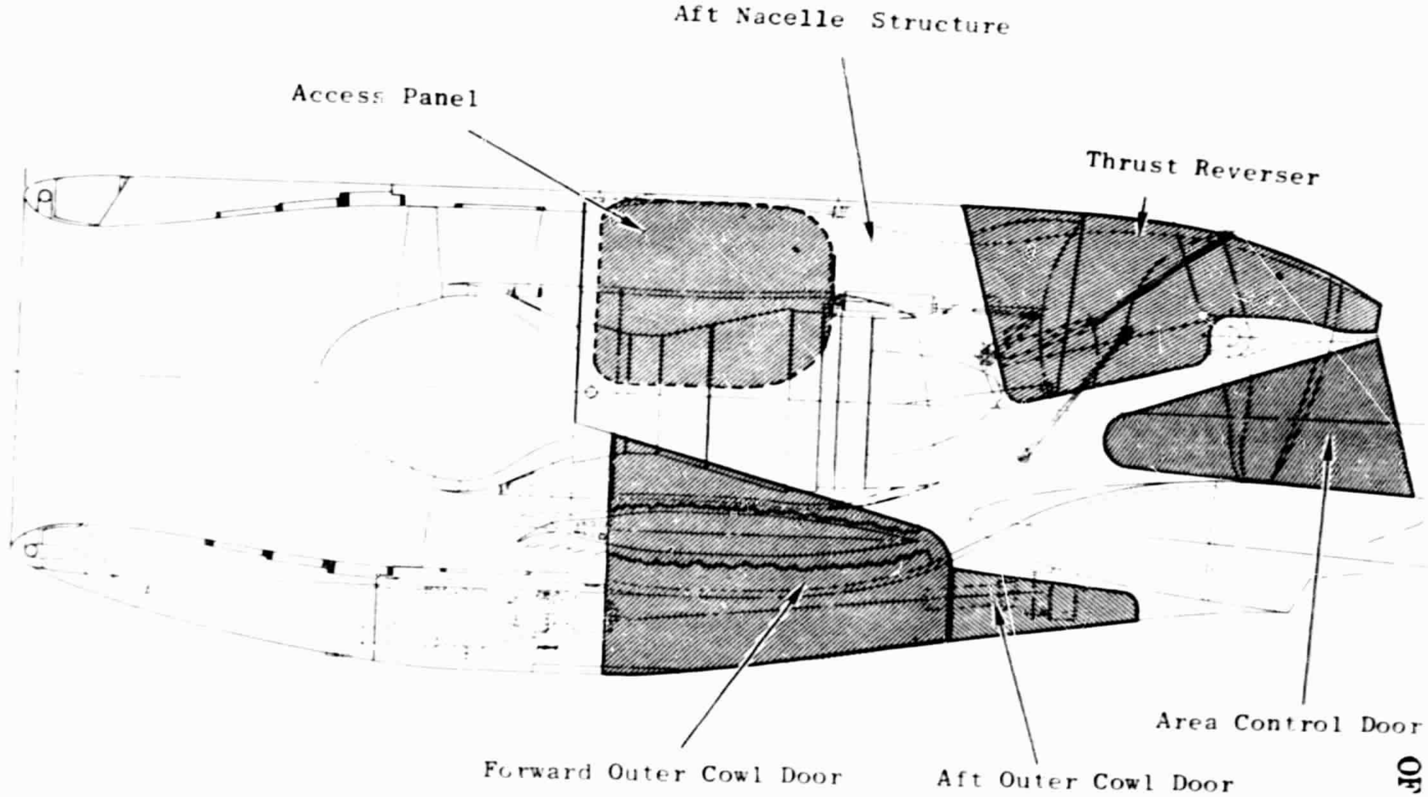

Figure 56. OTW Fan Bypass Duct Components.

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Table XI. Engine Mount and Aft Nacelle Structure Loads.

Location	Max Flight		5 Blades Out	
	N	lb	N	lb
Engine forward side mount	88,960 71,168	20,000 Axial 16,000 Vertical	194,378 400,320	43,700 Axial 90,000 Vertical
Engine forward top	64,496	14,500 Side	492,838	110,800 Side
Engine aft mount	53,821	12,100 Vertical	299,795	67,400 Vertical
Forward wing connection	84,512 64,496 32,248	19,000 Axial 14,500 Vertical 7,250 Side	938,350 460,235 246,419	210,960 Axial 103,470 Vertical 55,400 Side
Aft wing connection	4,626 28,467	1,040 Axial 6,400 Vertical	34,694 213,949	7,800 Axial 48,100 Vertical
Thrust reverser pivot	46,704 26,688	10,500 Axial 6,000 Vertical	46,704 26,688	10,500 Axial 6,000 Vertical

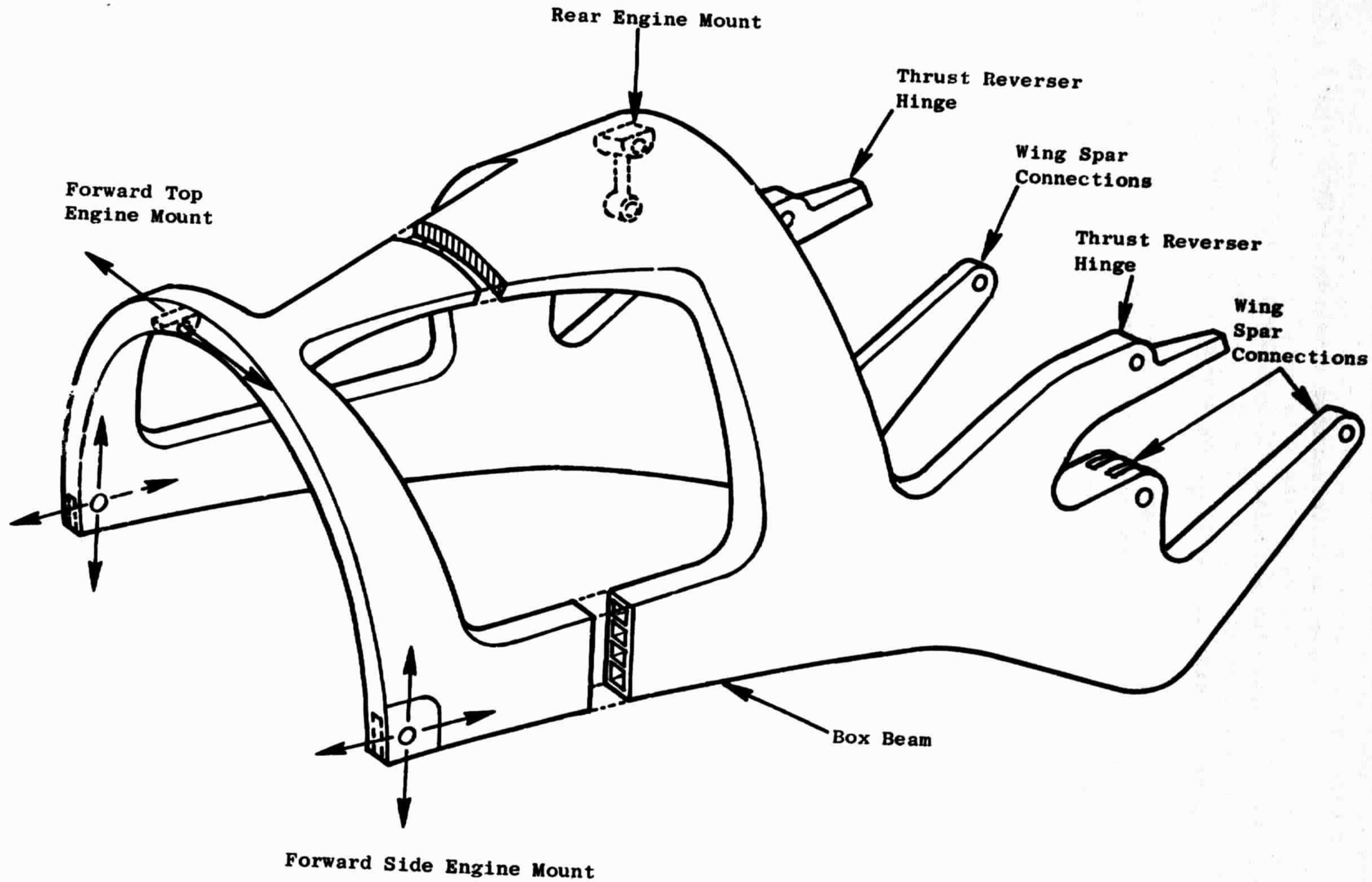


Figure 57. Aft Nacelle Structure.

titanium fittings integral with the mount box beam section at its forward end. These are pin-type connections which transfer engine vertical and axial loads (i.e., thrust, torque, yaw). The main box beam section is attached to the wing at the front and rear spars, with integrated titanium fittings again being utilized. The main load-transferring attachment is at the front spar and is of a multiple lug and clevis design which transmits side, axial, and vertical loads. The aft spar attachment is a titanium pin joint with side clearance so as to only resist the overturning moment of the mount structure.

Support for the thrust reverser clam shell blocker and the fan nozzle area control doors is provided by a box-beam-type appendage, which is integral with the mount structure and extends up and aft from the main box beam section just forward of the wing front spar attachment fitting. This appendage is of the same construction as the rest of the mount and contains an integral titanium reverser pivot support fitting and attachment points for the area control door hinges. An elastomer seal is installed around the forward face of the mount at the fan frame interface. This seal, in conjunction with the lower forward nacelle door seal, provides a continuous overboard air seal around the fan frame aft face at assembly.

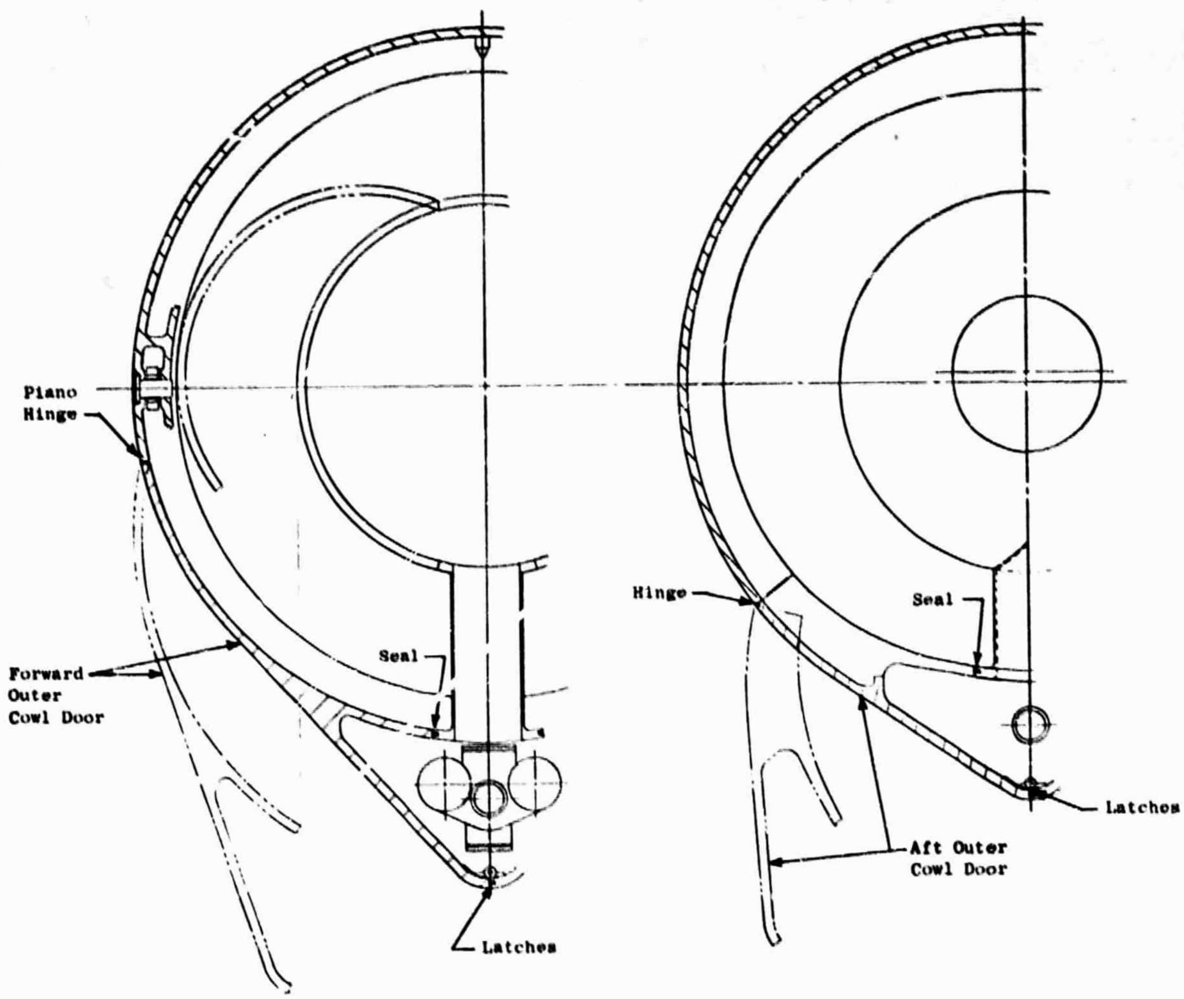
5.1.2.2 Nacelle Access Panels/Doors

For engine maintenance, removal, and installation, there are two access panels and four hinged doors located in the nacelle aft of the fan frame (see Figure 56).

The doors are located in the lower half of the nacelle between the fan frame and the aircraft wing leading edge. The larger forward doors extend from Station 207.5 to Station 273, providing access to the engine from the fan frame to the accessory gearbox area, while the smaller aft doors extend from Station 273 to Station 300, providing access to the gearbox area. These doors are hinged to the bottom of the engine-mount box beam sections and provide a continuation of the fan duct and nacelle outer surfaces, mating with the fan duct lower pylon on the fan duct (inner) surface and with the circumferentially adjacent door on the outer surface bottom centerline (see Figures 58 and 59). The doors are latched to each other at the outer surface bottom centerline and are mated to the adjacent structure/door by a radial tongue and groove joint which stabilizes and locates the door (see Figure 60). To prevent overboard leakage of the fan air, elastomer chevron-type seals are provided around the periphery of the doors.

Two access panels are located in the upper quadrants of the outer cowl, one on each side, in the enclosure formed by the engine mount structure box beam sections and partial rings.

The access panels are completely removable and are retained in place by quick-disconnect-type fasteners located around the edges of the panels. These panels are also sealed by chevron-type seals.



- Core Access
- Engine Removal

Figure 58. Fan Cowl Doors.

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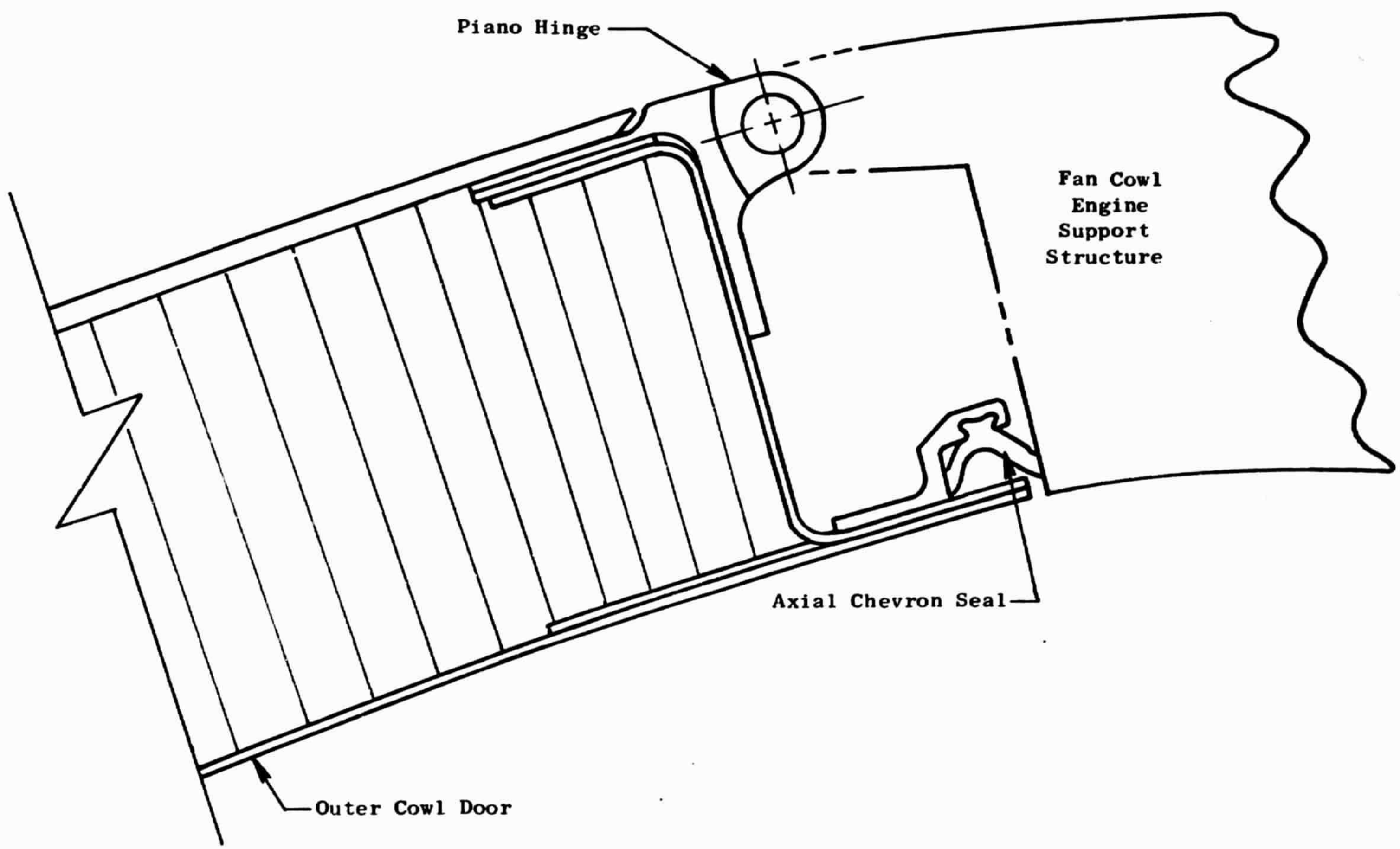


Figure 59. Fan Cowl Door Hinge and Seal.

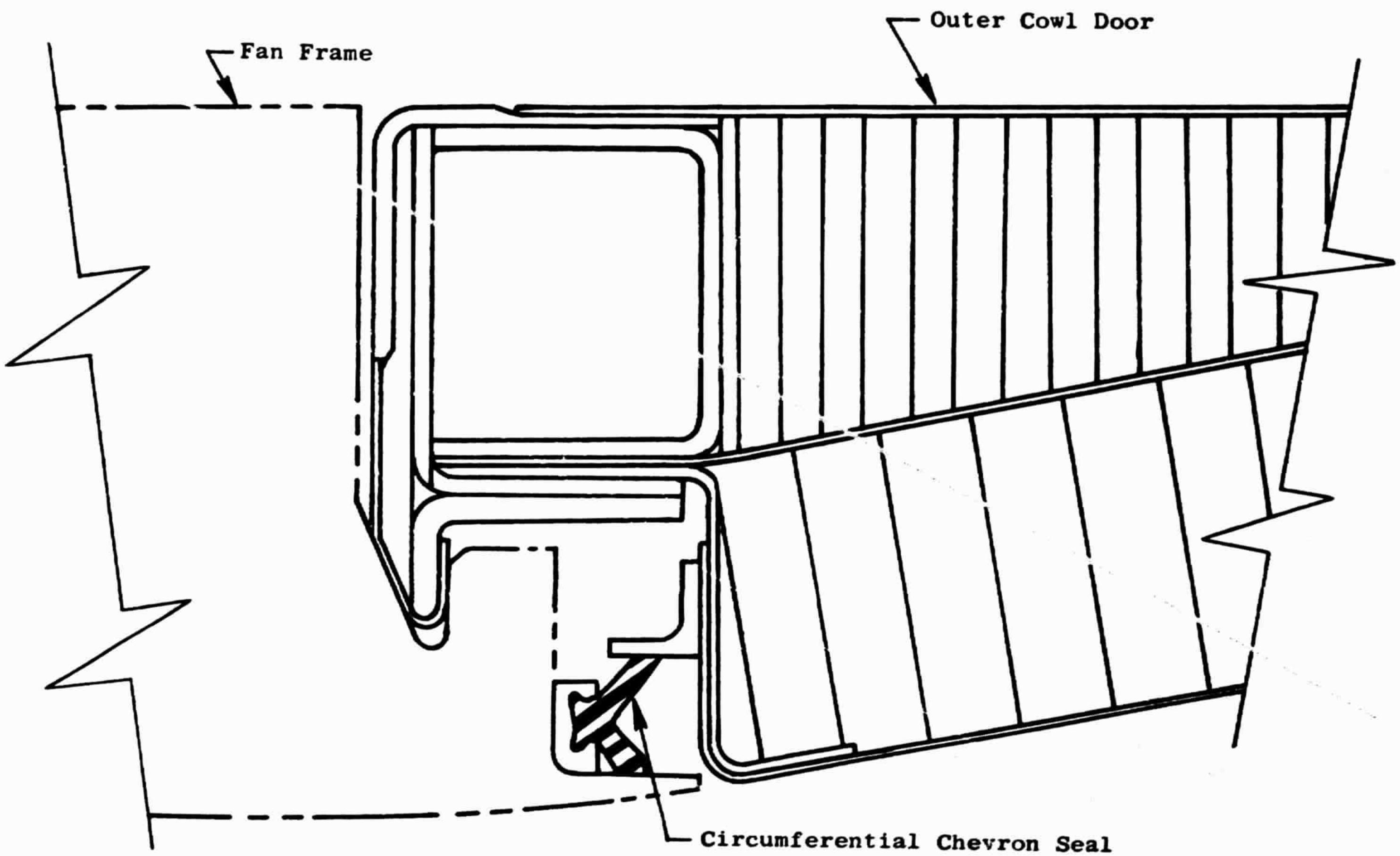


Figure 60. Outer Cowl/Fan Frame Attachement.

The construction of the panels and doors is of similar honeycomb sandwich construction. The acoustic treatment is integral with the inner skins, which are perforated to the required degree to meet the sound suppression requirements.

The honeycomb core is made from corrosion-resistant coated 5052 aluminum foil. To prevent any accumulation of fluids in the honeycomb sandwich interior, which could present an icing or corrosion problem, the individual cells of the door honeycomb are interconnected by slots providing continuous drainage paths to the lowest point of the door, where overboard drain holes are located.

5.1.3 Reverser and Area Control Doors

The target-type reverser system utilizes a single clam-shell blocker supported from the aft nacelle structure. In the stowed (forward thrust) position, the blocker, and the upper surface of the wing form a "D"-shaped engine exhaust nozzle. For reverse thrust, the clam shell is rotated until its trailing edge comes in close proximity to the upper wing surface, at the same time exposing an opening in the upper nacelle surface. The clam shell then acts as a large scoop and redirects the engine exhaust upward and forward through the exposed gap. (See Figure 22)

The primary blocker support is provided by a pivot bearing on each side. The blocker is actuated by mechanical jack screws, one on each side, attached to the outside edge of the blocker forward of the pivot. A translating lip (See Figure 49) is mounted on the blocker by means of a pivot bearing and connected to the fixed structure with a drag link. As the clamshell blocker is rotated aft by means of the jack screws, the drag link causes the lip extension to rotate relative to the blocker. In the reverse position, this movable lip extends the clamshell blocker and serves to turn the exhaust flow through a greater angle than the blocker alone would do, thereby increasing the amount of reverse thrust available from this system.

The reverser clamshell blocker itself is made up of two shells; the inner one is the pressure load carrying shell and is constructed of stainless-steel-clad aluminum sheet, reinforced by stainless steel hat sections. These materials are used because of the relatively high temperature ~617 K (~650° F) of this surface and also for its better erosion-resistant properties compared to composite materials. The outer shell forms a continuation of the outer nacelle surface and is a composite honeycomb sandwich, the skins being of a graphite/polyimide system and the core being fiberglass reinforced polyimide honeycomb. This outer shell is mechanically fastened at its forward edge to the metal inner shell. The other three edges have slip joints to allow for differential thermal expansion during operation. The loads in this configuration are shown in Table XII.

Table XII. Thrust Reverser System Loads.

	N	lb
Blocker pressure	133,400	30,000
Lip pressure	20,906	4,700
Lip pivot	8,896	2,000
Lip drag link	6,672	1,500
Blocker pivot	53,821	12,100
Actuator	48,928	11,000

An elastomer seal is provided along the sides and the forward edge of the inner shell to prevent overboard leakage of the exhaust gases during forward thrust operation.

The area control doors are located in the aft fixed-structure portion of the exhaust nozzle between the thrust reverser and the upper surface of the wing. They are triangular in shape, the apex being at the forward end. The aft edge forms a portion of the exhaust nozzle throat (see Figure 56). These doors are mounted to the fixed structure by means of three hinges along the upper edge of each door. Two of these hinges are powered-type hinges driven by flexible cables from a rotary hydraulic actuator. The construction of the doors is similar to that of the outer shell of the reverse blocker; specifically, graphite/polyimide skins bonded to a fiberglass-reinforced polyimide honeycomb core, with graphite/polyimide closeouts. These doors form the variable area nozzle during forward thrust operation (reverser stowed).

5.2 DIGITAL CONTROL

The control system for the flight version of the OTW propulsion system consists of a full authority digital control, a backup fuel and stator control, and appropriate flow control valves, actuator, and sensors. All system components are mounted on the propulsion system. A schematic of the system is shown in Figure 61.

The digital control manipulates three engine variables. Fuel flow is manipulated on a closed loop basis and exhaust nozzle area and core stators are scheduled to:

- Set percent of rated thrust
- Provide rapid thrust response

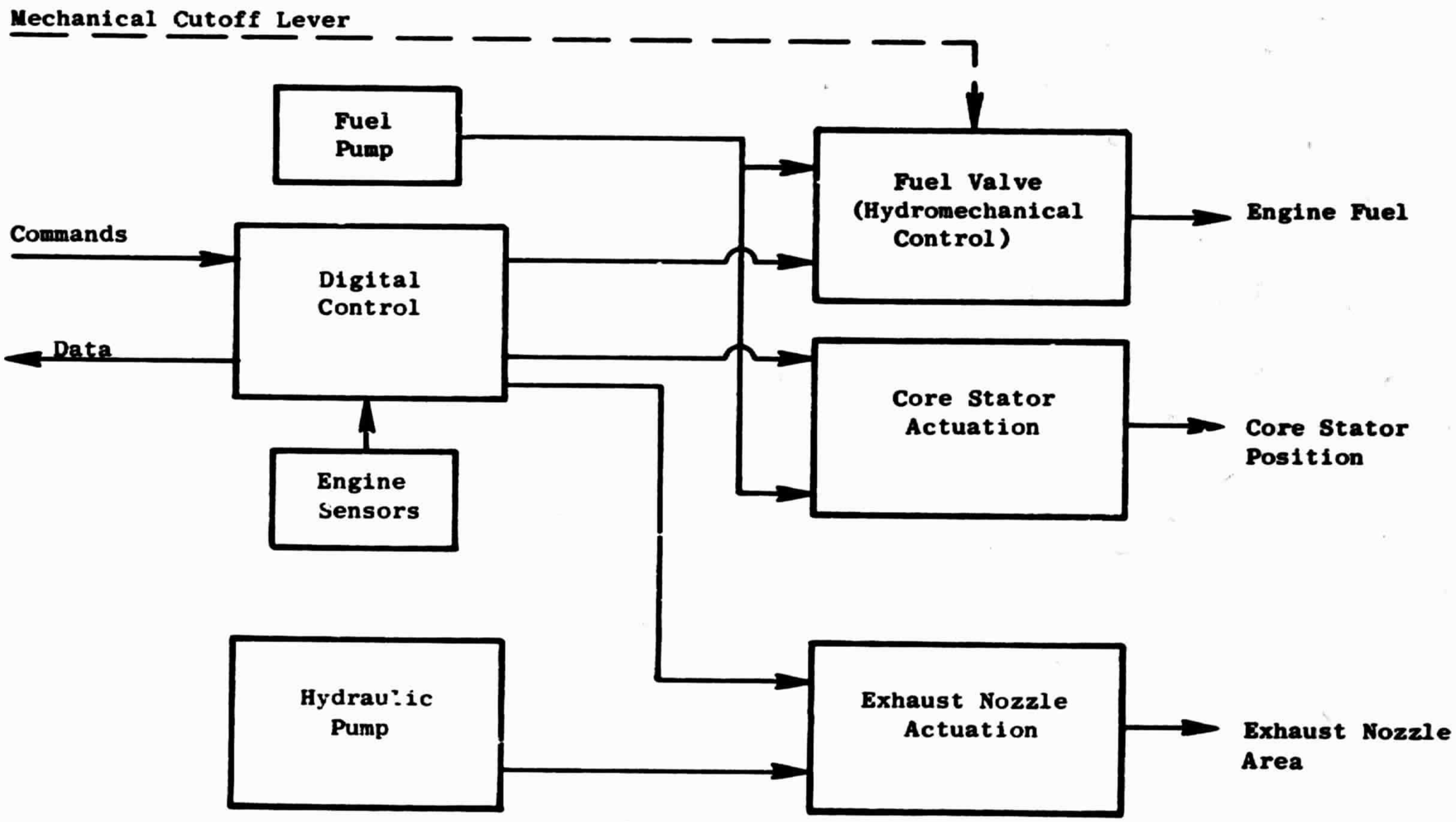


Figure 61. OTW Control System.

- Provide low idle thrust
- Reduce noise level
- Provide positive control of engine limits
- Reduce pilot workload.

Studies during the design of the experimental engine have established the sets of controlled and manipulated variables. These sets of variables will be utilized in the flight system after verification on the experimental engine. The prime set of controlled and manipulated variables includes:

- Percent of rated thrust is set through a fan speed demand which is controlled by modulating engine fuel flow.
- Exhaust nozzle area is scheduled as a function of power demand.
- Core stator angle is scheduled by corrected core speed.

5.2.1 System Operation

The overall system has many similarities with current systems except that all computation, during normal operation, is performed by an engine-mounted digital control and the power-command signal is transmitted electronically from the cockpit power lever to the engine digital control.

Actuation of the fuel shutoff lever in the cockpit drives a mechanical cable which opens the fuel stopcock on the engine-mounted fuel valve. The fuel stopcock is separated from the power lever to allow operation in the forward and reverse modes from a single power-command device. Mechanical actuation is chosen for stopcock actuation because it is a two-position operation where accuracy is not required and operational reliability is of prime importance.

Movement of the engine power levers in the cockpit by the pilot or the automatic throttle system will generate an electronic power-command signal to the digital control and a mechanical signal to the backup control. The electronic power-command signal is generated by a transducer integrated with the cockpit power demand system. The transducer is integrated with the throttle because throttle motions will facilitate pilot monitoring and pilot override capability is preserved. Cockpit power lever motion to the backup fuel valve may be transmitted mechanically or electrically depending on the results of a weight, reliability, accuracy, cost trade off study. The specific interface between the aircraft and the engine will be a splined shaft similar to current systems.

The electronic signal received by the digital control will be interpreted as a percent of rated thrust command. Each throttle position above flight idle will relate to a percentage of rated thrust. Thrust rating

could be in terms of maximum climb rating. Hence, positioning of the power lever angle at the takeoff position will be interpreted by the control as 100 plus percent of maximum climb rating. The control schedule and operating laws will be designed to provide the percent of rated thrust capability over the operating regime by sensing the environmental conditions. The engine control will return a signal to the cockpit to confirm the thrust command signal. As currently envisioned, the engine control will also incorporate the capability to receive an operating mode command. The operating mode signal could be integrated with the power lever console. The purpose of the operating mode signal is to allow the selection of a different set of control laws or engine rating for specific operating conditions (i.e., reverse, approach, etc.). For a particular application, this additional flexibility may not be required. The concept of incorporating the engine rating within the engine control and commanding percent of rated thrust as function of discrete power lever positions should result in a significant reduction in pilot workload.

Movement of the engine power levers in the cockpit from the idle position toward takeoff will result in the following events on the engine. Engine fuel flow, core speed, and fan speed will increase toward takeoff values. Final fan speed will be a corrected fan speed associated with takeoff thrust. Compressor core stators will move from closed to open as corrected core speed increases. As the aircraft climbs toward the cruise condition the exhaust nozzle will be closed from takeoff to cruise area as a function of environmental conditions (PTO, TT2-A8 schedule bias). To achieve the required rapid thrust response capability during approach conditions, the core stators will be reset in the closed direction as the power lever is retarded. This stator closure maintains a high core speed as the throttle is retarded and results in rapid thrust response with power lever advances. The core stator reset is automatically removed at high power settings.

5.2.2 Automatic Engine Limits

The digital control incorporates functions to automatically prevent the engine from exceeding design limits. The specific engine parameters which are protected by the digital control are low pressure turbine speed (LPT), core speed (N2), high pressure turbine inlet temperature (T4), and inlet Mach number (M_{11}). The LPT core speed limits are mechanized by sensing the value of these parameters and comparing to a reference limit within the digital control. If the sensed value attempts to exceed the limit, engine fuel flow is cut back to maintain operation at the limit. The turbine temperature limit is mechanized by computing the value of turbine inlet temperature from fuel flow, compressor discharge temperature, and pressure measurements. Engine fuel flow is retarded if the calculated temperature exceeds a reference value in the digital control to prevent operation above the temperature limit. If desired, this reference limit could be scheduled as a function of fan inlet temperature to extend the life of hot section parts. A computed turbine inlet temperature is being used to experimentally evaluate this concept. For the flight system, this computation concept or a

turbine blade optical pyrometer may be used. Inlet Mach number is computed from measurements of inlet static pressure and free stream total pressure, which are Mach number functions. The measured value is compared to a reference and fuel flow is retarded to prevent operation above the limit which would generate high inlet distortion. The control system is designed to allow stable, continuous operation on all of the limits. However, the engine is not expected to run on the speed or temperature limits during normal operation unless engine performance has deteriorated.

5.2.3 Safety Features

In addition to the above operational limits the digital control incorporates automatic fuel flow cutback features to prevent engine damage. These features are:

- Engine fuel flow is limited through a compressor discharge sensor and digital control logic to prevent compressor overpressure. In addition, this function limits fuel flow as a function of compressor discharge pressure to prevent damage from turbine overtemperature as a result of compressor surge.

- Engine fuel flow is automatically cutback to idle in the event the actual reverser position differs from the commanded position by a prescribed amount (i.e., inadvertent reverse).

- Engine fuel flow is automatically cut off if the rate of change of low pressure turbine speed exceeds a prescribed value or the level of low pressure turbine speed exceeds a prescribed value. This event could occur if the low pressure turbine load was lost.

5.2.4 Sensor Failure Protection

The digital control incorporates an advanced failure indication and corrective action (FICA) concept to prevent loss of operational capability in the event a control system sensor fails. Essentially, this concept consists of calculating the values of the sensed parameters from the engine inputs by use of a simplified model of the engine which is contained within the digital control memory. The calculated values of the sensed parameters are compared with the measured values and the model is updated to zero the error as long as the variation between the calculated and measured values do not exceed a prescribed amount.

If the prescribed amount is exceeded on a given parameter, the measured parameter is replaced with the calculated value and the system continues operating normally. If a sensor is replaced with a calculated value, a signal is sent to the cockpit noting that a specific control sensor has failed. Maintenance action would be required prior to the next flight. Incorporation of this concept will prevent damage or shutdown and allow

normal operation with a control system sensor failure. The concept allows use of the digital control's inherent computational capability and will be more cost effective than redundant sensors.

5.2.5 Backup Control

Because the reliability of a single-channel, full-authority digital control has not been demonstrated, and because a reliability level greater than 100,000 hours mean time between engine shutdowns would probably be required for commercial service, a backup or reversionary control is considered necessary for initial service. A backup control can be implemented in several ways, namely:

- Redundant digital control with separate sensors
- Redundant analog control with separate sensors
- A simplified hydro-mechanical control.

General Electric has studied the backup control problem and has tentatively selected a simplified hydro-mechanical control for the backup control approach.

Recent studies by General Electric on a two-variable engine have shown that a simple feed-forward control approach can provide backup control. These studies have shown that both main fuel flow and compressor stators are functions of WF/PTO, which is scheduled as a function of power lever angle. Transient protection is provided by controlling the time rate of change of fuel flow and core stators by internally regulating the rate of change of power lever angle. This concept can be readily adapted to work in conjunction with the components of a digital control system.

The interface between the digital control and the backup control is a fail-fixed servovalve (FFSV). This device is an electrohydraulic servovalve which has been modified to temporarily lock the servovalve output stage if the electrical input fails to zero or hardover in either direction. With this device incorporated, the following events will occur with a digital control failure:

- Fuel flow will be initially held at the level at time of failure and then over a period of several minutes drift upward until it intersects the level required to satisfy the WF/PTO schedule set by the backup fuel control power lever. Engine power may be retarded or advanced by movement of the cockpit power lever. Transient fuel flow will be scheduled by rate of change of power lever angle.

- The fan nozzle area will be initially held at the area at time of failure and then over a period of several minutes drift to an area between cruise and takeoff where the load on the actuators is balanced.

- The core compressor stators will initially be locked in the position held at the time of failure and then over a period of several minutes will drift closed to a reduced performance point. Stators will be opened and closed with power level changes by a power lever schedule to provide safe operation with reduced performance.

With the above features, the propulsion system will provide essentially normal power immediately after a digital control failure. As the variable geometry drifts to zero load positions, engine power may be reduced. Transient performance will be reduced and limited to safe rates of change by the backup control. Pilot monitoring of fan speed and turbine temperature will be required with this failure condition at high power levels.

5.2.6 Engine Condition Monitoring

The digital control will serve as an accumulator and processor of engine condition information. The specific data list has not been finalized but will probably consist of rotor speed, fuel flow, engine vibration, geometry positions, core engine pressures and temperatures, and lube system temperature and pressure. This data will be transmitted to the aircraft via a multiplex link. It is planned that the digital control will incorporate an indelible read-write memory which will compute and store information associated with operational hours, life cycle counts, overtemperature events, and high vibrations, etc. During engine servicing this memory would be interrogated by ground support equipment to define required maintenance action. In addition to the information processed by the digital control, the following parameters will be sent to the cockpit on separate hardwired cables to allow engine operation in the event a failure occurs in the multiplex link or the digital control: low pressure turbine speed, low pressure turbine discharge temperature, and lube oil pressure.

5.2.7 Aircraft Interface

The interface between the aircraft propulsion control system and the engine control is a critical item and requires detail coordination and design by aircraft and engine manufacturers to achieve an optimum configuration. Preliminary studies in this program have led to the conclusion that there will be a mechanical and electrical interface.

The mechanical interfaces will be a splined shaft which actuates the fuel stopcock with movement of the fuel shutoff lever in the cockpit and a splined shaft which actuates the backup control power lever.

The electrical interface will be at the engine digital control. This interface will receive power command, operating mode, and air data information from the aircraft, and will transmit engine condition data to the aircraft.

5.3 LUBE/FUEL SYSTEM

QCSEE bearings and seals components, other than the main reduction gear, can operate throughout the flight map with oil supply temperature of 422 K (300° F) and oil scavenge temperatures as high as 450 K (350° F). Local bearing temperatures rise above 450 K (350° F) during some portions of a flight. These components generate 68,288 J/sec (3880 Btu/min) at takeoff power.

The AISI 9310 gears and bearing outer races in the main reduction gear have 422 K (300° F) metal temperature limits at all conditions. These components require a lube system with the ability to cool the oil below 367 K (200° F) during most flight conditions. This system must accommodate an additional 111,672 J/sec (6345 Btu/min) from the gearbox during takeoff.

Fuel system requirements are:

- 323 K (122° F) supply max at all conditions
- Fuel heating to 273 K (32° F) at filter inlet with 233 K (-40° F) soak and 225 K (-55° F) flight.

The following system was selected in order to meet QCSEE requirements.

- Fuel recirculation to aircraft fuel tanks
- Split bypass return fuel to A/C tanks - returning portion of lube heat to A/C tanks
- Ambient cooling of A/C wing surfaces sufficient to avoid tank temperature rise
- Priority arrangement of reduction gearbox and main lube heat exchangers
- Combined function oil-to-fuel heating for filter ice protection.

Results of the OTW flight study are summarized here. The selected system, with slight additional tuning, will meet all system requirements.

- Two CF6-size fuel/oil heat exchangers, 11.8 kg (26 lb) engine dry weight, aluminum removable core - nonbrazed
- Supplemental oil cooling (recirculation) at cruise and idle descent only
- No fuel tank overheating

- 427 K (309° F) AGMA scoring temperature (takeoff). 424 K (304° F) AGMA scoring temperature (climb)

- Fuel heating capability (de-ice) 233 K (-41° F) soak
  219 K (-65° F) flight

The system shown in the OTW fuel/oil schematic (Figure 62) was selected to meet QCSEE requirements. Oil discharged from the lube supply pump follows two parallel paths. One path directs oil to all the normal lube system components. The other path routes oil through a supplemental cooler prior to entering the main reduction gear. Thus, only the oil required for the reduction gear is cooled to lower temperature levels. All scavenge oil is routed through a common heat exchanger before returning to the oil tank.

The heat study conditions as shown in Table XIII were selected from the November 1, 1974 QCSEE Technical Requirements and the July 18, 1974 Curtis-Wright Gearing Data.

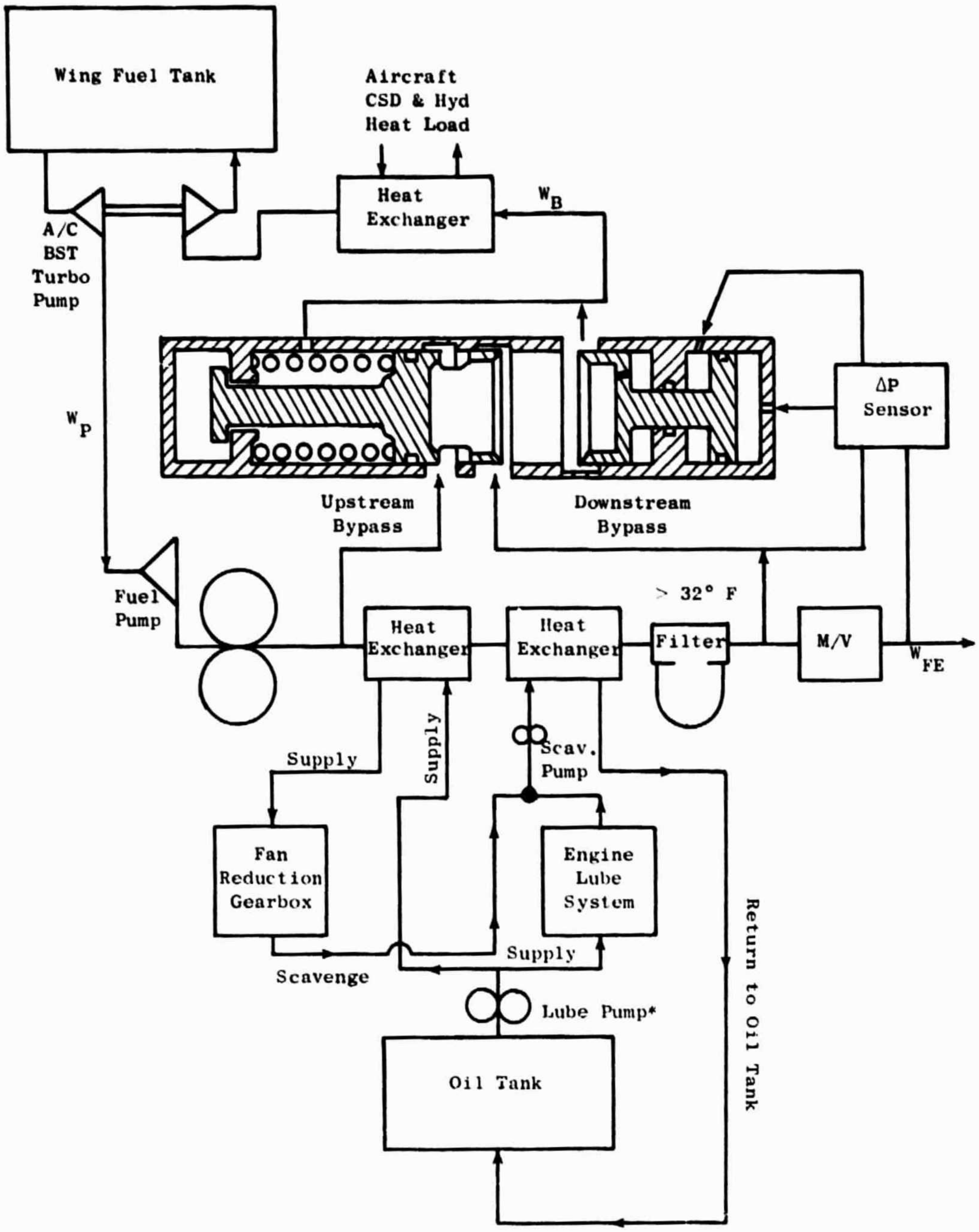
Heat loads from the reduction gear and engine lube system have been calculated for each condition. Table XIV summarizes these results.

Fuel and oil flows are tabulated (Table XV) for the same operating conditions. While upstream bypass fuel is returned to the tanks during the entire mission, heated downstream bypass fuel (after the heat exchanger) is only returned during cruise and descent.

During the previously tabulated heat rejection rates and fuel/oil flows a fuel tank temperature rise rate was calculated in Table XVI for each condition in the flight map. It was assumed that 13.9 m^2 (150 ft^2) of wing cooling surface was available for each engine. During all conditions except cruise and descent, the only heat returned to the tanks is from the fuel pump.

Calculated reduction gear bearing and gear temperatures are near their 422 K (300° F) limits. As seen in Table XVI, maximum calculated metal temperatures never exceed 427 K (309° F).

A final design objective was to assure that under cold fuel conditions, the fuel entering the filter would be at least 273 K (32° F). As can be seen in Table XVII, the system meets the design objectives of 233 K (-40° F) ground soak and 225 K (-55° F) in-flight cool-down.



* Includes metering elements to assume known flow split at all operating conditions.

Figure 62. OTW Fuel/Oil Schematic.

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Table XIII. Heat Study Conditions.*

Condition	Case No.	Altitude	Flight Mach No.	Reduction Gear Efficiency, %	LPT		Fan Speed, %	Core Speed, %
					kW	hp		
Ground Idle	111	SL	0	98.63	75	100	15.1	55.4
Takeoff	1	SL	0	99.11	12,526	16,798	101.5	96.3
Climb	301	SL	0.38	99.10	12,193	16,351	94.5	95.4
Cruise	404	7.6 km (25K ft)	0.7	99.07	6,529	8,756	96.4	91.8
Descent	503	4.6 km (15K ft)	0.6	98.73	1,593	2,136	62.4	77.6
Approach	7	61 m (200 ft)	0.12	99.15	6,718	9,009	81.8	87.1
Max. Fan Torque	926 (revised)	SL	0	99.11	15,032	20,158	97.8	97.4

*From November 1, 1975 Technical Requirements and September 5, 1974 C-W Data Selected on the Basis of Max. AGMA Scoring ΔT.

Table XIV. Predicted Heat Loads and Oil Flows.

Condition	Reduction Gearbox		Engine Lube		Total Q		Oil* Flow	
	J/sec	Btu/min	J/sec	Btu/min	J/sec	Btu/min	kg/sec	lb/min
Ground Idle	1,021	58	5,984	340	7,005	398	1.34	178
Takeoff	111,672	6345	68,288	3880	179,960	10,225	2.34	310
Climb	109,912	6245	62,339	3542	172,251	9,787	2.32	307
Cruise	60,825	3456	56,373	3203	117,198	6,659	2.23	295
Descent	20,258	1151	24,482	1391	44,739	2,542	1.89	250
Approach	57,200	3250	41,782	2374	98,982	5,624	2.12	280
Max Fan Torque	134,006	7614	68,253	3878	202,259	11,492	2.37	313

* Density ≈ 886.8 kg/m^3 (7.4 lb/gal)

Table XV. Fuel and Oil Flows.

Condition	Engine Metered Fuel, W_{FE}		Engine Supply Fuel, W_P		Total Tank Return Fuel, W_B		Heat Exch. Tank Return Fuel		Reduction Gear Oil *		Engine Lube Oil *	
	kg/sec	lb/hr	kg/sec	lb/hr	kg/sec	lb/hr	kg/sec	lb/hr	kg/sec	lb/min	kg/sec	lb/min
Ground Idle	0.083	660	1.40	11,135	1.32	10,475	0	0	0.937	124	0.408	54
Takeoff	0.898	7126	2.44	19,356	1.54	12,230	0.032	257	1.63	216	0.711	94
Climb	0.919	7292	2.42	19,175	1.50	11,883	0.032	257	1.62	214	0.703	93
Cruise	0.472	3746	2.32	18,451	1.85	14,705	0.240	1910	1.56	206	0.673	89
Descent	0.161	1274	1.97	15,597	1.80	14,323	0.240	1910	1.32	174	0.575	76
Approach	0.531	4217	2.20	17,507	1.67	13,290	0.032	257	1.48	196	0.635	84

* Density = 886.8 kg/m^3 (7.4 lb/gal)

Table XVI. 50° C (122° F) Fuel Tank Temperature Rise Rate, Std + 31° F Day.

Condition	Temp Heat Exch. Fuel Inlet		Temp Heat Exch. Fuel Disch.		Temp Return Fuel To Tank		Total Return Q To Tank		Fuel Tank** Ambient Cooling		Fuel Tank** Temp Rise Rate		Scav Return Oil		Lube Supply Oil		Red. Gear Supply Oil		Bearing† Race		AGMA† Scoring Factor	
	K	F	K	F	K	F	J/sec	Btu/min	J/sec	Btu/min	K/sec	F/min	K	F	K	F	K	F	K	F	K	F
Ground Idle	324	123	364	195	324	123	2,922	166*	1,408	80	0.00019	0.02	345	161	338	149	338	148	342	155	340	152
Takeoff	324	124	417	290	326	127	14,696	835	1,408	80	0.00139	0.15	443	337	396	253	363	194	419	294	427	309
Climb	324	124	411	279	325	127	14,238	809	22,582	1624	-0.00148	-0.16	425	306	394	250	361	190	414	286	424	304
Cruise	324	124	403	265	334	142	46,341	2633	94,547	5372	-0.00491	-0.53	417	291	377	218	358	184	395	251	397	255
Descent	324	123	372	219	331	136	31,786	1806	76,102	4324	-0.00454	-0.49	378	220	357	183	350	170	368	202	366	198
Approach	324	124	408	274	326	127	12,725	723	25,450	1446	-0.00130	-0.14	412	282	381	226	362	192	404	268	404	268

* Fuel Pump 'Q' Only - no Lube Heat
** Per Engine - 4649 kg (10,250 lb) Fuel - 13.9 m^2 (150 ft^2) Lower Wing Surface Cooling
† 9310 Steel 422 K (300 ° F) Limit.

Table XVII. Fuel Heating Capability.

Condition	Engine Fuel Supply*		Fuel Filter Inlet	
	K	° F	K	° F
Ground Idle	233	-41	273	32
Takeoff	180	-136	273	32
Climb	186	-125	273	32
Cruise	194	-111	273	32
Descent	219	-65	273	32
Approach	189	-120	273	32

* Design Objectives: 233 K (-40° F) Ground Soak
≈225 K (-55° F) In-flight Cooldown

5.4 PROPULSION SYSTEM WEIGHT

The OTW flight propulsion system is projected to meet the following:

Uninstalled thrust 93,408 N (21,000 lb)

Uninstalled weight 1324 kg (2919 lb)
Fn/Wt(u) = 70.5 N/kg (7.19 lb/lb)

Installed thrust 90,294 N (20,300 lb)

Installed weight 1943 kg (4284 lb)
Fn/Wt (i) = 46.5 N/kg (4.74 lb/lb)

The above projection is based on experimental engine thrust and weight (to be measured during the QCSEE Program), with suitable adjustments in weight to account for material and design changes that have been identified as applicable to the flight system. A breakdown of the flight propulsion system weight projection is provided in Table XVIII.

Uninstalled weight includes, in addition, the following:

- High Mach suppressed inlet
- Fan duct doors
- Core cowl
- Aft nacelle

- Nozzle/reverser doors and linkage
- Nozzle/reverser actuation
- Oil cooler
- Instrumentation
- Drains and vents

Normally aircraft furnished components such as starter, aircraft accessory gearbox, bleed piping, fire detection and extinguishing system, and engine mounts are excluded. The aft nacelle includes only internal and external skins, with the structure (corresponding to a pylon) considered aircraft furnished.

Table XVIII. OTW Flight Propulsion System Weight.

Nacelle Components	Equivalent Flight Weight	
	kg	lb
Inlet	156.5	345
Fan Duct Doors	95.3	210
Core Cowl	43.1	95
Core Exhaust	38.1	84
Aft Nacelle Skins	113.4	250
Nozzle/Reverser Doors and Linkage	120.7	266
Nozzle/Reverser Actuation	25.9	57
Oil Cooler	18.1	40
Instrumentation	5.4	13
Drains and Vents	2.3	5
Total Installation	619.2	1356
Engine	1324.1	2919
Total Propulsion System	1943.3	4284

6.0 PROPULSION SYSTEM PERFORMANCE

6.1 OTW FLIGHT ENGINE

Engine performance data were generated for the QCSEE over-the-wing flight engine for use in evaluation of the flight, economic, and acoustic characteristics of commercial, short-haul aircraft. Engine performance over a range of flight conditions is shown in Table XIX. (For title definitions, see Acronyms and Symbols in Table XX.) The performance levels shown for the flight engine are based on component characteristics projected for the certification level engine.

The system performance in Table XIX includes the following installation factors:

- Ram recovery
- Customer bleed (Boeing estimated requirement)
- Customer shaft power extraction (Boeing estimated requirement)
- Conical exhaust nozzle

Ram recovery utilized in generating installed performance is shown in Figure 63. The inlet throat is sized to produce 0.79 Mach number at a corrected flow of 405.5 kg/sec (894 lb/sec).

Customer bleed levels shown in Table XIX were set to match Boeing requirements. Because of the strong effect of compressor bleed, additional data were also generated at reduced bleed levels (Table XXI) to allow evaluation of bleed effects on performance. Shaft power extraction was held constant at 59,656 W (80 hp) for all flight conditions.

The performance levels shown in Tables XIX and XXI are based on a conical exhaust nozzle with a 0.995 velocity coefficient. Specific effects of application of a "D"-type nozzle were estimated by Boeing for their particular applications.

Table XIX. QCSEE Over-the-Wing Flight Engine Installed Performance.

Alt m	Alt ft	XM	DTAMB K	DTAMB °F	FN N	FN lb	sfc g/sN	sfc (lb/hr)/lb	W1 kg/sec	W1 lb/sec	BPR	WB27 kg/sec	WB27 lb/sec
Takeoff													
0	0	0	273	+31	89,934	20,219	0.010	0.346	386	851	10.21	0	0
		0.1	273	+31	77,747	17,479	0.011	0.395	390	860	10.32		
		0.2	273	+31	66,773	15,012	0.013	0.450	399	879	10.57		
Max Climb													
3,048	10,000	0.4	256	0	40,472	9,099	0.016	0.571	290	640	9.34	0.9	2.0
		0.5			38,813	8,726	0.018	0.619	313	689	9.65		
		0.6			37,243	8,373	0.019	0.666	337	742	9.95		
		0.7			35,842	8,058	0.020	0.714	363	801	10.26		
4,572	15,000	0.4			34,734	7,809	0.016	0.568	245	541	9.28		
		0.5			33,654	7,566	0.017	0.614	264	583	9.55		
		0.6			32,693	7,350	0.019	0.656	285	629	9.80		
		0.7			31,834	7,157	0.020	0.700	308	680	10.04		
Max Cruise													
6,096	20,000	0.6	256	0	26,190	5,888	0.018	0.645	236	521	9.84	0.9	2.0
		0.7			25,456	5,723	0.020	0.690	255	563	10.09		
		0.8			24,890	5,596	0.021	0.741	276	608	10.32		
7,620	25,000	0.6			22,338	5,022	0.018	0.639	197	434	9.76		
		0.7			21,982	4,942	0.019	0.679	213	470	9.93		
		0.8			21,724	4,884	0.021	0.726	230	507	10.11		
9,144	30,000	0.6			18,646	4,191	0.018	0.636	162	358	9.69		
		0.7			18,557	4,172	0.019	0.673	176	387	9.83		
		0.8			18,539	4,168	0.020	0.716	190	419	9.96		
10,668	35,000	0.6			15,510	3,487	0.018	0.634	132	292	9.66		
		0.7			15,421	3,467	0.019	0.669	143	316	9.78		
		0.8			15,443	3,472	0.020	0.720	155	342	9.86		
12,192	40,000	0.6			11,356	2,753	0.018	0.635	105	231	9.71		
		0.7			12,196	2,742	0.019	0.669	113	250	9.83		
		0.8			12,196	2,742	0.020	0.712	122	270	9.93		

Alt m	Alt ft	XM	T27 K	T27 °F	P27 N/cm²	P27 psia	T16 K	T16 °F	P16 N/cm²	P16 psia	T56 K	T56 °F	P56 N/cm²	P56 psia	T8 K	T8 °F	P8/P0	A18 cm²	A18 in.²
Takeoff																			
0	0	0	---	---	---	---	336	145	13.3	19.3	935	1,223	12.4	18.0	396	252	1.287	17,155	2,659
		0.1	---	---	---	---	336	145	13.4	19.4	928	1,211	12.4	18.0	394	249	1.293	17,155	2,659
		0.2	---	---	---	---	337	146	13.5	19.4	921	1,198	12.5	18.1	392	244	1.306	17,155	2,659
Max Climb																			
3,048	10,000	0.4	542	515	48.3	70.3	307	92	10.4	15.1	881	1,126	9.7	14.4	366	198	1.447	14,187	2,199
		0.5	548	527	51.6	74.8	311	99	11.0	15.9	880	1,124	10.4	15.1	368	202	1.546	14,206	2,202
		0.6	555	539	54.9	79.6	316	108	11.7	16.9	881	1,125	11.0	15.9	371	207	1.640	14,239	2,207
		0.7	563	553	58.6	85.0	322	119	12.5	18.1	882	1,128	11.6	16.8	375	215	1.755	14,290	2,215
4,572	15,000	0.4	569	492	40.0	57.9	297	74	8.6	12.5	877	1,118	8.3	12.0	357	182	1.483	14,187	2,199
		0.5	536	504	42.5	61.7	301	81	9.1	13.2	876	1,116	8.7	12.6	356	185	1.566	14,213	2,203
		0.6	542	516	45.6	66.1	306	90	9.7	14.1	876	1,116	9.2	13.3	362	191	1.666	14,252	2,209
		0.7	550	530	49.0	71.1	312	101	10.4	15.1	877	1,118	9.7	14.1	366	199	1.787	14,303	2,217
Max Cruise																			
6,096	20,000	0.6	525	485	35.6	51.7	293	68	7.9	11.4	844	1,060	7.4	10.7	347	164	1.660	14,252	2,209
		0.7	527	489	38.4	55.7	299	78	8.5	12.3	846	1,062	7.9	11.4	351	172	1.780	14,297	2,216
		0.8	536	504	41.6	60.3	306	90	9.2	13.3	848	1,067	8.9	12.3	356	181	1.926	14,368	2,224
7,620	25,000	0.6	505	449	28.2	40.9	283	50	6.4	9.34	840	1,053	6.1	8.82	337	167	1.682	14,258	2,210
		0.7	513	464	30.8	44.6	289	60	7.0	10.1	841	1,054	6.5	9.43	342	155	1.809	14,310	2,218
		0.8	522	479	33.6	48.7	296	72	7.5	10.9	843	1,058	7.0	10.18	347	165	1.963	14,361	2,226
9,144	30,000	0.6	489	421	21.4	31.1	272	30	5.2	7.56	838	1,049	4.9	7.15	327	129	1.700	14,264	2,211
		0.7	498	436	23.7	34.4	278	41	5.6	8.15	838	1,048	5.3	7.67	332	137	1.833	14,316	2,219
		0.8	506	451	26.3	38.1	284	52	6.1	8.86	839	1,050	5.7	8.38	337	147	1.992	14,368	2,227
10,668	35,000	0.6	475	395	16.6	24.1	262	12	4.2	6.06	838	1,048	4.0	5.75	318	112	1.721	14,271	2,212
		0.7	482	408	17.9	25.9	267	21	4.5	6.54	837	1,047	4.3	6.17	322	119	1.855	14,329	2,221
		0.8	489	421	19.5	28.3	273	32	4.9	7.11	836	1,045	4.6	6.68	327	128	2.017	14,374	2,228
12,192	40,000	0.6	469	385	13.2	19.1	259	7	3.3	4.77	839	1,050	3.1	4.53	315	107	1.722	14,277	2,213
		0.7	477	398	14.1	20.5	265	17	3.6	5.15	838	1,048	3.4	4.86	319	115	1.850	14,329	2,221
		0.8	484	412	15.2	22.1	271	28	3.9	5.60	839	1,050	3.6	5.26	324	124	2.018	14,374	2,228

• 0.06 MW (80 hp) shaft power extraction at all conditions.

Table XX. Performance Nomenclature

Alt	Geopotential Pressure Altitude, m (ft)
XM	Flight Mach Number
DTAMB	Ambient Temperature Minus Standard Atmosphere Temperature, K (°F)
FN	Net Thrust, N (lb)
sfc	Specific Fuel Consumption, g/SN (lb/hr/lb)
W1	Engine Inlet Total Airflow, kg/sec (lb/sec)
BPR	Bypass Ratio (Bypass Stream Flow/Core Stream Flow)
WB27	Core Compressor Interstage Bleed Flow, kg/sec (lb/sec)
TB27	Core Compressor Interstage Bleed Temperature, K (°F)
PB27	Core Compressor Interstage Bleed Pressure, N/cm^2 (psia)
T16	Bypass Stream Total Temperature at Plane of Confluence, K (°F)
P16	Bypass Stream Total Pressure at Plane of Confluence, N/cm^2 (psia)
T56	Core Stream Total Temperature at Plane of Confluence, K (°F)
P56	Core Stream Total Pressure at Plane of Confluence, N/cm^2 (psia)
T8	Exhaust Flow Average Mixed Temperature, K (°F)
P8/PO	Exhaust Nozzle Exit Total to Static Pressure Ratio
AE8	Effective Exhaust Nozzle Exit Area, cm^2 ($in.^2$)

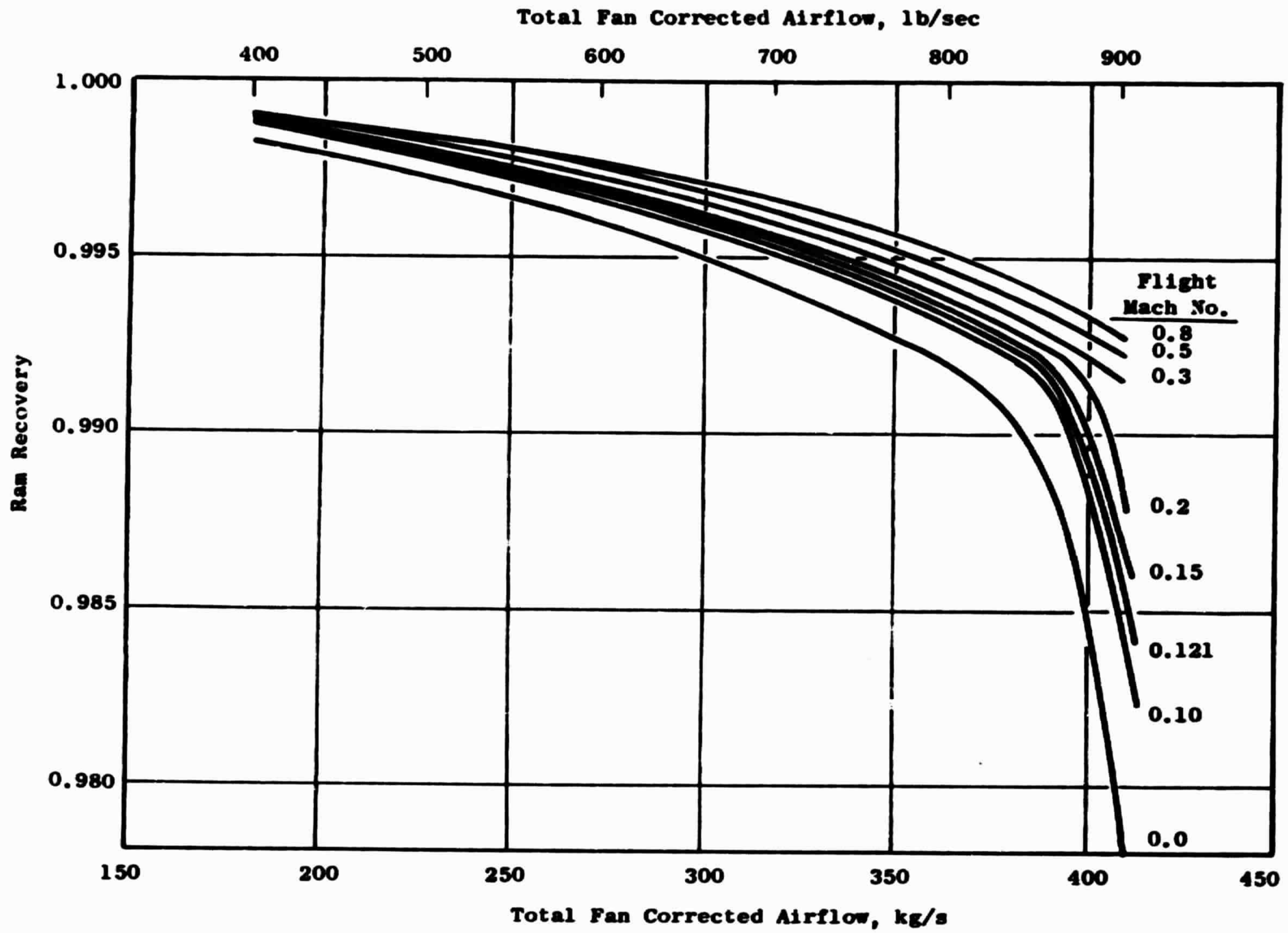

Figure 63. Ram Recovery Characteristics.

Table XXI. QCSEE Over-the-Wing Flight Engine Installed Performance.

Alt m	Alt ft	XM	ΔTamb K	ΔTamb °F	Fn N	Fn lb	sfc g/N·s	sfc (lb/hr)/lb	W1 kg/sec	W1 lb/sec	BPR	WB27 kg/sec	WB27 lb/sec
Max Cruise													
6,096	20,000	0.6	256	0	27,223	6,120	0.018	0.634	238	525	9.86	0.5	1.0
↓	↓	0.7	↓	↓	26,461	5,949	0.019	0.677	257	566	10.11	↓	↓
		0.8			25,036	5,629	0.021	0.727	277	611	10.36		
7,620	25,000	0.6			23,365	5,253	0.018	0.627	199	438	9.76		
↓	↓	0.7			23,001	5,171	0.019	0.665	215	473	9.94		
		0.8			22,729	5,110	0.020	0.711	231	510	10.11		
9,144	30,000	0.6			19,638	4,415	0.018	0.623	164	361	9.71		
↓	↓	0.7			19,553	4,396	0.019	0.650	177	390	9.86		
		0.8			19,540	4,393	0.020	0.699	191	422	9.96		
10,668	35,000	0.6			16,240	3,651	0.018	0.622	133	294	9.56		
↓	↓	0.7			16,275	3,659	0.019	0.654	144	318	9.77		
		0.8			16,435	3,695	0.020	0.691	156	345	9.85		
12,192	40,000	0.6			12,623	2,838	0.018	0.627	105	232	9.71		
↓	↓	0.7	↓	↓	12,677	2,850	0.019	0.658	114	251	9.82	↓	↓
		0.8			12,788	2,875	0.020	0.696	123	272	9.92		
Throttle Curve													
9,144	30,000	0.72	256	0	19,549	4,395	0.019	0.666	180	396	9.86	0.5	1.0
					18,570	4,175	0.019	0.663	178	393	9.98		
					16,618	3,736	0.019	0.665	175	386	10.29		
					14,665	3,297	0.019	0.671	171	378	10.67		
↓	↓		↓	↓	9,777	2,198	0.020	0.722	161	356	11.92	↓	↓
					2,932	659	0.061	1.464	169	372	15.97		

Alt ft	XM	T27 K	T27 °F	P27 N/cm²	P27 psia	T16 K	T16 °F	P16 N/cm²	P16 psia	T56 K	T56 °F	P56 N/cm²	P56 psia	T8 K	T8 °F	P8/P0	A18 cm²	A18 in.²
Max Cruise																		
20,000	0.6	522	480	39.4	57.2	294	70	7.9	11.5	842	1,056	7.5	10.9	348	167	1.675	14,258	2,210
	0.7	534	501	42.1	61.1	300	80	8.5	12.4	844	1,059	8.0	11.6	353	175	1.795	14,303	2,217
	0.8	542	515	45.2	65.5	307	92	9.2	13.4	847	1,064	8.5	12.4	359	186	1.948	14,353	2,225
25,000	0.6	514	465	32.3	46.9	284	52	6.5	9.44	838	1,049	6.2	8.96	339	150	1.700	14,264	2,211
	0.7	521	478	34.8	50.4	290	62	7.0	10.16	839	1,050	6.6	9.55	343	158	1.820	14,316	2,219
	0.8	529	492	37.5	54.4	296	73	7.6	11.02	841	1,054	7.1	10.31	349	168	1.982	14,368	2,227
30,000	0.6	499	439	26.1	37.8	274	33	5.3	7.65	835	1,043	5.0	7.26	329	133	1.721	14,277	2,213
	0.7	507	453	28.1	40.8	279	43	5.7	8.25	835	1,043	5.4	7.79	336	145	1.856	14,329	2,221
	0.8	515	467	30.5	44.3	286	56	6.2	8.97	838	1,049	5.8	8.43	340	152	2.016	14,374	2,228
35,000	0.6	484	412	20.6	29.9	263	14	4.2	6.13	833	1,039	4.0	5.86	320	116	1.761	14,286	2,214
	0.7	493	427	22.4	32.5	269	24	4.6	6.62	832	1,038	4.3	6.28	326	127	1.880	14,335	2,222
	0.8	500	441	24.4	35.4	275	35	5.0	7.21	833	1,039	4.7	6.80	329	133	2.047	14,387	2,230
40,000	0.6	475	396	15.3	22.2	261	9	3.3	4.81	835	1,043	3.2	4.57	317	110	1.735	14,277	2,213
	0.7	484	411	16.8	24.4	266	19	3.6	5.20	834	1,041	3.4	4.90	321	118	1.875	14,335	2,222
	0.8	492	426	18.4	26.7	272	30	3.9	5.66	835	1,043	3.7	5.36	326	127	2.041	14,383	2,229
Throttle Curve																		
30,000	0.72	509	456	28.6	41.5	280	45	5.8	8.39	835	1,043	5.5	7.96	335	143	1.886	14,335	2,222
		502	444	27.9	40.5	279	43	5.7	8.28	816	1,009	5.4	7.78	331	136	1.862	14,329	2,221
		491	423	26.5	38.4	277	38	5.6	8.07	787	956	5.2	7.52	324	124	1.833	14,380	2,230
		479	403	24.9	36.1	276	36	5.4	7.85	761	909	5.0	7.28	318	112	1.764	14,290	2,215
		460	367	20.5	29.7	268	22	5.0	7.31	697	795	4.6	6.63	303	86	1.649	14,239	2,207
		420	296	15.2	22.1	261	10	4.5	6.50	596	612	3.7	5.32	280	45	1.443	17,155	2,659

* 60 kW (80 hp) shaft power extraction at all conditions.

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7.0 ACOUSTICS

Noise predictions were made for the Boeing 914.4 m (3000 ft) runway aircraft employing four QCSEE OTW engines. The predictions were based on aircraft data (trajectory, flap angles, power settings) supplied by Boeing and corresponding General Electric engine cycle data. Noise objectives and measuring points are shown in Figure 64.

7.1 OTW NACELLE ACOUSTIC DESIGN

To meet the 95 EPNdB goal, the following engine noise suppression levels are necessary:

- Forward radiated fan noise - 13 ΔPNdB suppression
- Aft radiated fan noise - 9.0 ΔPNdB suppression
- Core exhaust noise - no suppression

The required inlet suppression level is obtained with a treated high Mach inlet having a throat Mach number of 0.79. Acoustic suppression is supplied by the wall treatment during approach when the reduced power setting results in lower inlet throat Mach numbers. The estimated inlet suppression during approach is 8.0 ΔPNdB.

The required fan exhaust suppression is obtainable with acoustic treatment on the walls only; an acoustically treated splitter is not required in the fan exhaust as used in the experimental engine. The acoustic features of the nacelle designed to meet these suppression requirements is shown in Figure 65.

7.2 SYSTEM NOISE LEVEL PREDICTIONS

At various points along the Boeing-supplied flight path, predictions of the total system (engine + jet/flap) noise levels were made, employing the procedures outlined in Appendix I to the QCSEE Statement of Work. Tables XXII and XXIII summarize the predictions for the 152.4 m (500 ft) sideline, and are representative of the peak sideline noise levels to be measured on takeoff and approach for the Boeing aircraft. Suppression levels typical of the nacelle shown in Figure 65 were assumed. The peak 152.4 m (500 ft) sideline suppressed levels are seen to be 95.4 EPNdB on takeoff and 88.8 EPNdB on approach.

The predicted system noise levels along various sideline distances, along with those for a direct flyover, were used to generate the estimated noise contours for 90, 95, and 100 EPNdB. These contours are presented in Figure 66. The peculiar shape of these contours is due to the assymetric character of the noise sources, particularly the jet/flap noise, which peaks directly

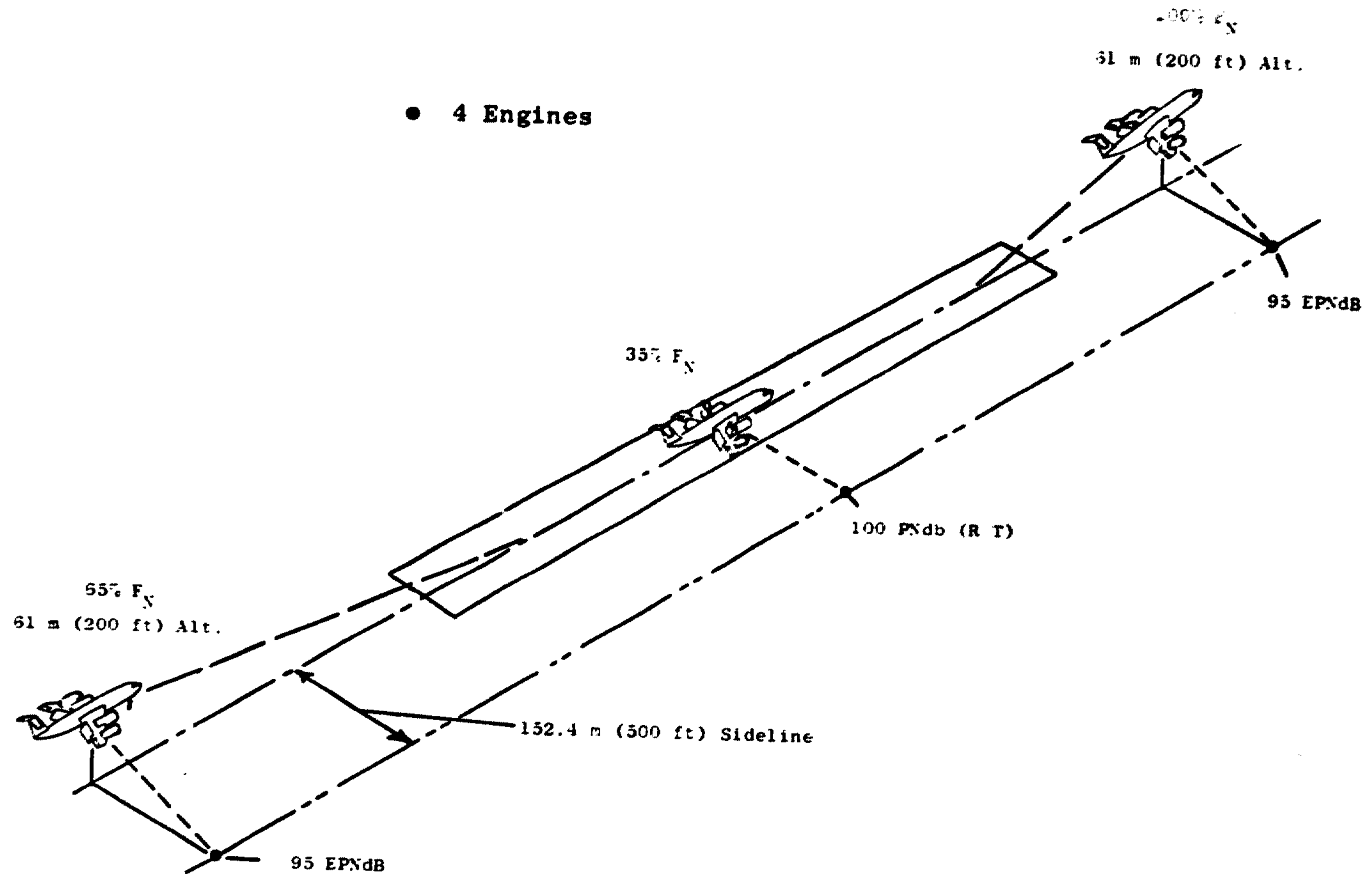

Figure 64. Acoustic Requirements.

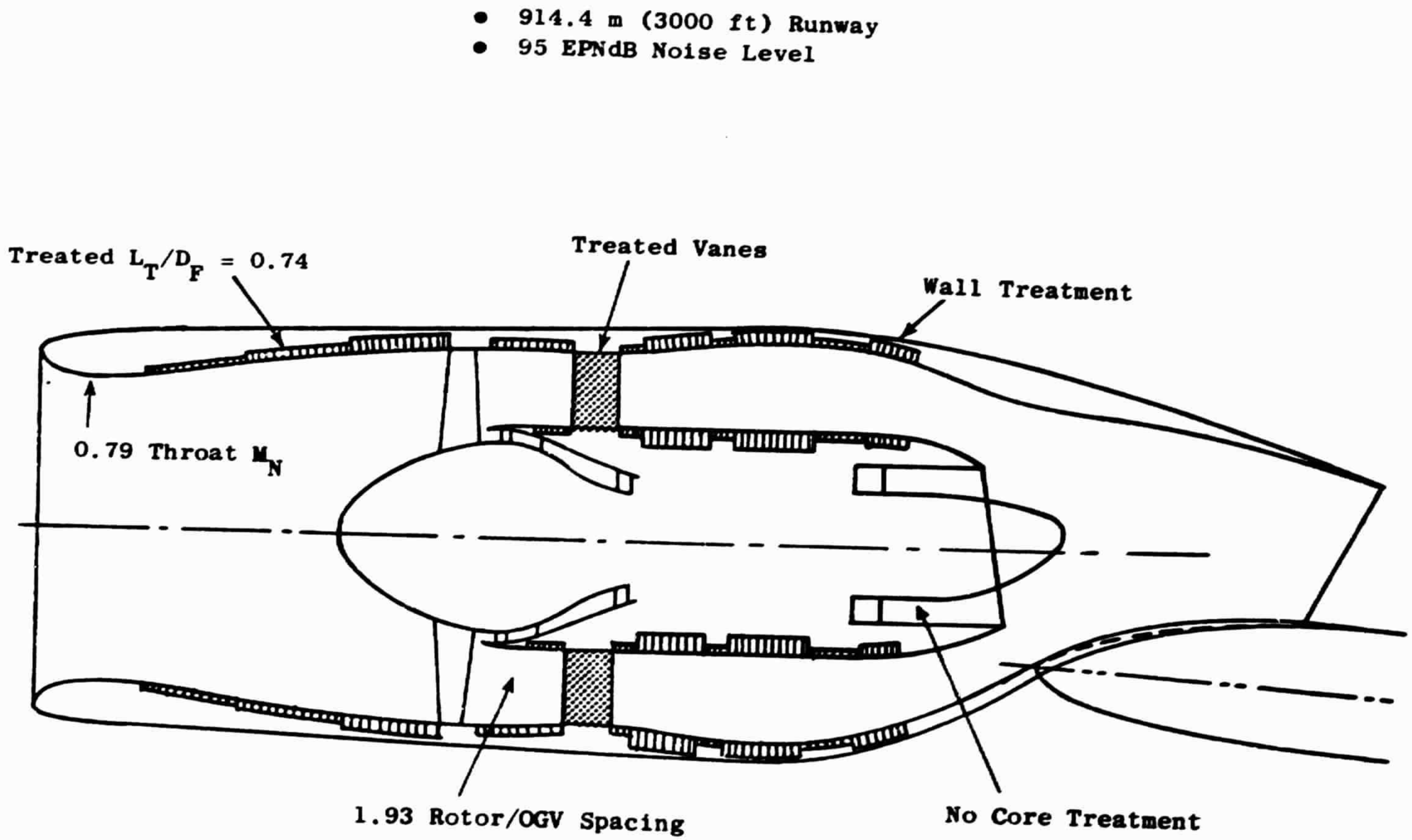

Figure 65. OTW Engine.

Table XXII. QCSEE OTW 914.4 m (3000 ft) Runway Peak Noise Levels.

- Takeoff Power [F_N = 86,158 N (19,370 lb)]
- 152.4 m (500 ft) Sideline, 61 m (200 ft) Altitude
- Aircraft Velocity = 58.4 m/sec (113.5 Knots)
- Blown Flap Angle = 0 Degrees

	Max Fwd Angle (90°) PNdB				Max Aft Angle (120°) PNdB			
	Fan	Turbine	Combustor	J/Flap	Fan	Turbine	Combustor	J/Flap
Single Engine Unsuppressed 61 m (200 ft) Sideline	115.4	97.9	95.8	105.6	115.7	100.6	101.1	103.1
Corrections - Appendix I Proc. Steps 1 - 5 152.4 m (500 ft) Sideline @ 61 m (200 ft) Alt	-10.1	-11.8	-9.0	-8.1	-11.1	-13.2	-9.8	-9.1
"View Angle" Correction	--	--	--	-3.5	--	--	--	-3.5
Step 6 No. Engines Fuselage Shielding Dirt/Grass Ground	 +6.0 -1.2 -1.5	 +6.0 -1.2 -1.5	 +6.0 -1.2 -.5	 +6.0 -1.2 -.5	 +6.0 -1.2 -1.5	 +6.0 -1.2 -1.5	 +6.0 -1.2 -.5	 +6.0 -1.2 -.5
Step 7 Inlet Cleanup	-2.0	--	--	--	--	--	--	--
Step 8 Jet/Flap Relative Velocity	--	--	--	-3.5	--	--	--	-3.5
Step 9 Wing Shielding	--	-5.0	-3.5	--	-5.0	-5.0	-3.5	--
Total Corrections	-8.8	-13.5	-8.2	-10.8	-12.8	-14.9	-9.0	-11.8
Corrected Level	106.6	84.4	87.6	94.8	102.9	85.7	92.1	91.3
Suppression	-13.0	--	--	--	-9.0	--	-	--
Suppressed Levels	93.6	84.4	87.6	94.8	93.9	85.7	92.1	91.3
Step 10 Sum Constituents	Max Fwd PNL 98.5				Max Aft PNL 98.2			
Step 11 PNdB to EPNdB	EPNdB 95.4							

Table XXIII. QCSEE OTW 914.4 m (3000 ft) Runway Peak Noise Levels.

- Approach Power (65% F_N)
- 152.4 m (500 ft) Sideline 61 m (200 ft) Altitude
- Aircraft Velocity = 47.8 m/sec (93 Knots)
- Blown Flap Angle = 40 Degrees

	Max Fwd Angle (90°) PNdB				Max Aft Angle (120°) PNdB			
	Fan	Turbine	Combustor	J/Flap	Fan	Turbine	Combustor	J/Flap
Single Engine, Unsuppressed 61 m (200 ft) Sideline	105.0	89.3	88.1	96.[illegible]	108.3	96.6	93.5	92.6
Corrections - App I Procedure Steps 1 - 5 152.4 m (500 ft) Sideline @ 61 m (200 ft) Alt	-10.1	-11.8	-9.0	-8.1	-10.8	-12.9	-9.5	-8.8
"View Angle" Correction	--	--	--	-3.5	--	--	--	-3.5
Step 6 No. Engines Fuselage Shielding Dirt/Grass Ground	 +6.0 -1.2 -1.5	 +6.0 -1.2 -1.5	 +6.0 -1.2 -.5	 +6.0 -1.2 -.5	 +6.0 -1.2 -1.5	 +6.0 -1.2 -1.5	 +6.0 -1.2 -.5	 +6.0 -1.2 -.5
Step 7 Inlet Cleanup	-3.5	--	--	--	--	--	--	--
Step 8 Jet/Flap Relative Velocity	--	--	--	-2.2	--	--	--	-2.2
Step 9 Wing Shielding	--	-5.0	-3.5	--	-5.0	-5.0	-3.5	--
Total Corrections	-10.3	-13.5	-8.2	-9.5	-12.5	-14.6	-8.7	-10.2
Corrected Level	94.7	75.8	79.9	86.7	95.8	82.0	84.8	82.4
Suppression	-8.0	--	--	--	-9.0	--	--	--
Suppressed Levels	86.7	75.8	79.9	86.7	86.8	82.0	84.8	82.4
Step 10 Sum Constituents	Max Forward PNL 90.9				Max Aft PNL 91.1			
Step 11 PNdB to EPNdB	EPNdB 88.8							

Contour Areas, Acres

Contour	Takeoff	Approach	Total
90	414	81	495
95	140	19	159
100	53	2	55

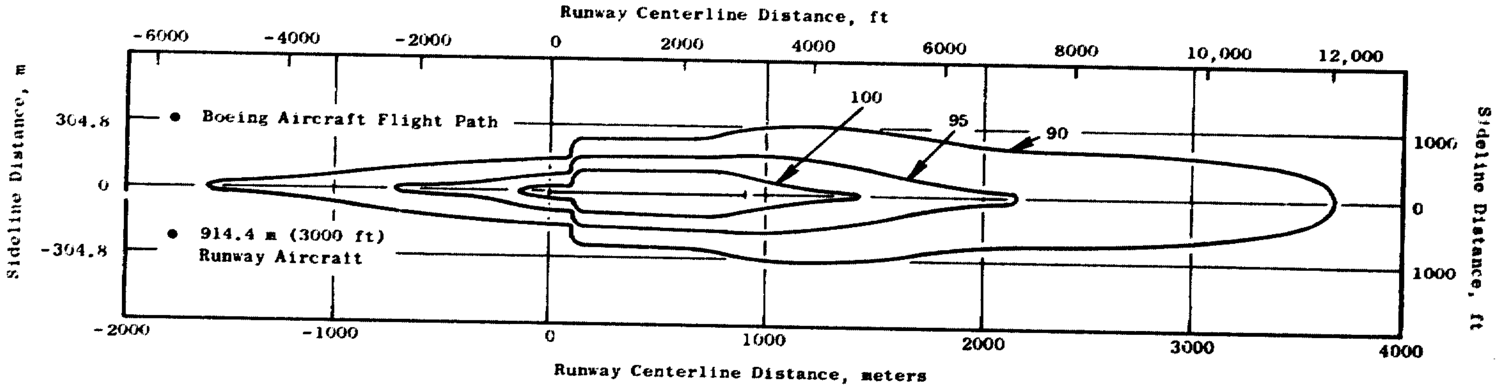

Figure 66. OTW Flight Engine Noise Contours.

underneath the aircraft and falls off fairly rapidly with increasing sideline distance. No attempt was made to optimize the flight path for noise requirements; flap angles, power settings, and climb gradients are held constant throughout (see Section 2.2 for definition of flight path). The use of cutback, two-part glide slopes, etc., whenever possible, might result in even further reductions to the already impressively low footprint areas.

8.0 ECONOMICS

The operating costs presented in this section were determined from formulae developed at Boeing to represent airline costs incurred in typical short-haul operations. These formulae were derived from historical costs experienced by the local service carriers in the United States. This group of air carriers was selected as being the most representative of a dedicated short-haul operator. The formulae provide one method to assess the comparative economic suitability of aircraft designed specifically to operate in the short-haul environment.

The historical cost element data were obtained from the data base of the CAB Form 41 as compiled in "An Airline Analytical Information System" (Reference 3). This program of data was originally developed by American Airlines and the data contained have been found to be reasonably reliable and consistent.

8.1 BASELINE AIRCRAFT

The development of the direct operating costs for the baseline aircraft was based on the assumptions listed in Table XXIV. The five categories which make up the direct operating costs are:

- Flight crew pay (the baseline aircraft is designed for a two man crew; cabin attendants are considered indirect cost).
- Fuel
- Maintenance (engines and airframe including labor and materials)
- Depreciation
- Insurance

The direct operating cost formulae have been computerized and require that both block time and block fuel be input as slopes and intercepts.

The direct operating cost of the baseline aircraft is presented in Figure 67 as a function of range. This cost, expressed in cents per available seat statute mile, shows the typical decline with range until the design range of 925 km (500 N Mi) is obtained with the maximum payload. Beyond this range, payload must be off loaded and the costs increased due to the reduction in available seats.

Tables XXV and XXVI present detailed breakdowns of the direct operating costs when determined by different costing formulae. The current (1975) version of the Boeing short-haul formula is an updated version of the 1974

Table XXIV. Data for Economic Analysis.

Block Fuel: 748 kg (1650 lb) + 5.6 kg/km (20.0 lb/st mi)

Block Time: 18 min + 0.076 min/km (0.123 min/st mi)

Boeing 1975 STOL Economic Rules and Dollars

Utilization: 2555 Block hr/yr

2-Man Crew

Fuel: $79.26/m^3$ (30¢ Per U.S. Gallon)

Study Prices: (1975 Dollars)

- Airframe: $19,112,000 (Includes $403,000 per Nacelle)
- Engine: $1,140,000 Each
- Spares: 6% Airframe: 30% Engines

Depreciation: 14 Years to 2% Residual Value

Insurance: 1% of Flyaway Price

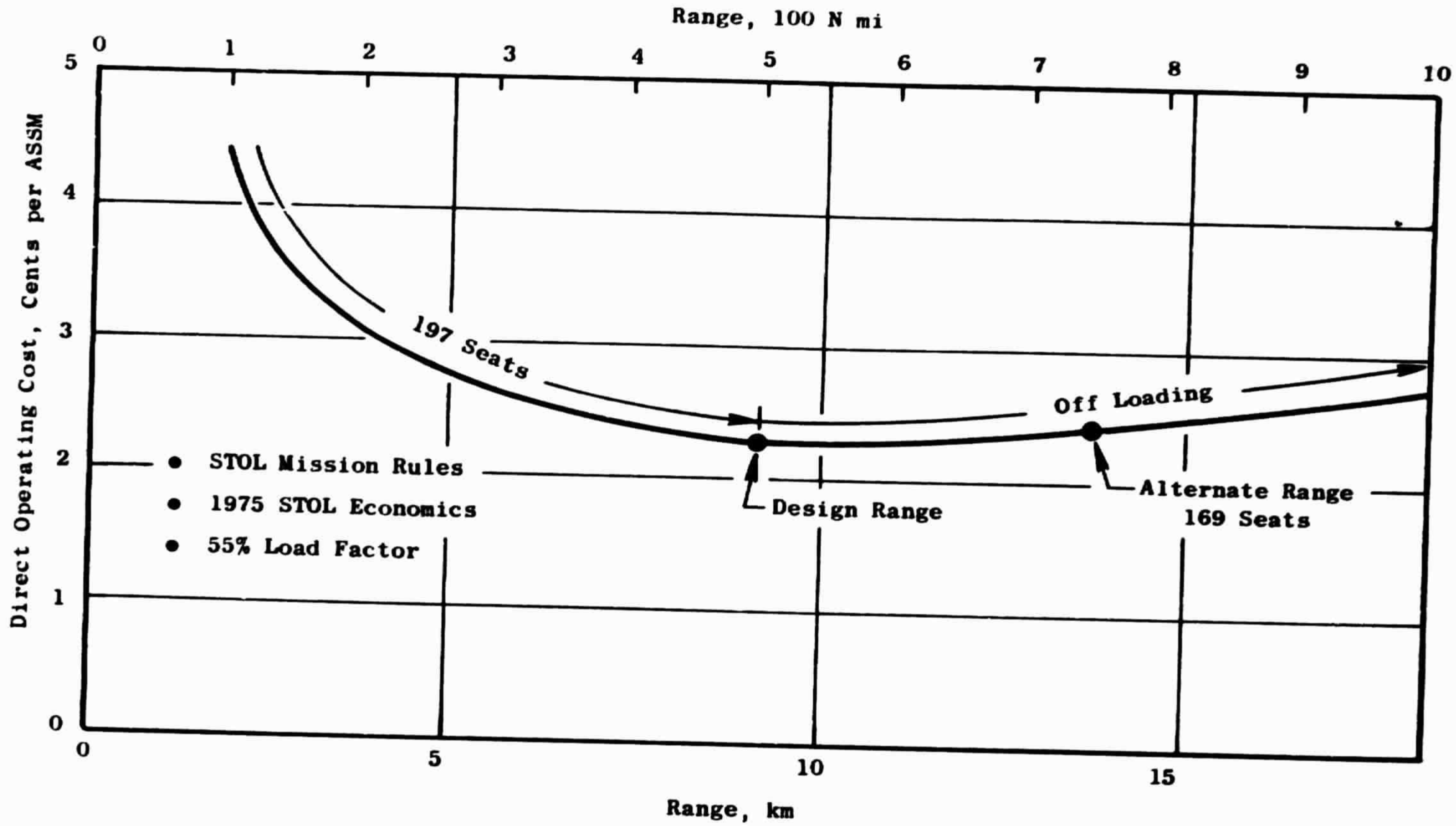

Figure 67. Direct Operating Cost.

Table XXV. Impact of One Years Inflation on Costs.

No.	Cost Item	Boeing 1974 QCSEE Short Haul Costs (1974 Dollars)	Boeing 1975 QCSEE Short Haul Costs (1975 Dollars)
1.	Flight Crew	231.51	247.67
2.	Fuel $79.26/m^3 (30¢/Gallon)	605.15	605.15
3.	Insurance	115.75	137.01
4.	Depreciation	897.78	1,060.92
5.	Airframe Dir. Labor/Cycle	35.64	38.35
6.	Airframe Dir. Labor/Hr	45.32	48.76
7.	Airframe Mat'l/Cycle	41.49	49.15
8.	Airframe Mat'l/Hr	31.31	36.85
9.	Engine Dir. Labor/Cycle	7.24	7.79
10.	Engine Dir. Labor/Hr	31.46	33.82
11.	Engine Mat'l/Cycle	27.75	31.07
12.	Engine Mat'l/Hr	104.51	119.14
13.	Maint. Burden	162.11	182.11
	Total DOC	2,337.02	2,597.79

Notes:
- Trip Cost in Dollars
- 926.5 km (500 N Mi) Range
- 2555 Hr/Yr Utilization

Table XXVI. Cost Comparison Between Carrier Types.

No.	Cost Item	Boeing 1975 CTOL	Boeing 1975 Short Haul
1.	Flight Crew	289.47	247.67
2.	Fuel \$79.26 m^3 (30¢/Gallon)	605.15	605.15
3.	Insurance	137.03	137.01
4.	Depreciation	1,061.10	1,060.92
5.	Airframe Dir. Labor/Cycle		38.35
6.	Airframe Dir. Labor/Hr.	73.99	48.76
7.	Airframe Mat'l/Cycle		49.15
8.	Airframe Mat'l/Hr.	88.17	36.85
9.	Engine Dir. Labor/Cycle		7.79
10.	Engine Dir. Labor/Hr.	39.83	33.82
11.	Engine Mat'l/Cycle		31.07
12.	Engine Mat'l/Hr.	168.83	119.14
13.	Maint. Burden	370.82	182.11
	Total DOC	2,834.39	2,597.79

Notes:
- Trip Costs in Dollars
- 926.5 km (500 N Mi) Range
- 2555 Hr/Yr Utilization

formula and accounts primarily for the impact of inflation during the year. Table XXV compares the direct operating costs of the baseline aircraft using the two short-haul formulae. Table XXVI presents a comparison of the direct operating costs of the baseline airplane as determined by the standard (CTOL) costing formula and the 1975 short-haul formula. Note that the CTOL formula yields a higher direct operating cost, which reflects the historical inherent differences between the trunk and local service carrier.

Operating cost sensitivities are presented in Figure 68 for variations in utilization, fuel, airframe, and engine prices. Increasing the utilization reduced direct operating costs at the design range by increasing the total yearly flights over which the years depreciation and insurance values may be allocated. This is a nonlinear relationship as shown in the curve on Figure 68. Increasing fuel prices impact the direct operating costs by about a one-to-four ratio. That is, increasing the fuel price by 10 percent results in a 2.4 percent increase in direct operating costs. Increases in airframe price of 10 percent will increase the cost by 4.1 percent. A 10 percent increase in engine price increases the costs by 1.9 percent. In the case of both the engine and airframe, the increase in direct operating costs occurs in the depreciation, insurance, and maintenance areas.

The cost sensitivities described above may be applied to the DOC shown in Figure 67 at the 925 km (500 N Mi) design range.

8.2 DESIGN TRADES

Trade studies were conducted to show the impact of cruise altitude and Mach number on direct operating costs and fuel utilization. The 914.4 m (3000 ft) field length design condition was preserved by maintaining 90040 kg (198,500 lb) takeoff gross weight and 167.7 m^2 (1805 ft^2) wing area. This results in a design trade between aircraft fuel and passenger payload capability.

The maximum number of passengers which can be transported to the 925 km (500 N Mi) stage length was parametrically determined as a function of cruise altitude and Mach number as shown on Figure 69. The weight of fuel burn shown on Figure 70 plus reserve fuel was parametrically traded for passenger payload, seating provisions, and a corresponding portion of the fuselage structural weight. Thus, the fuselage was considered variable to accommodate the passenger payload. This results in an operational empty weight variation as shown on Figure 71. The maximum altitude shown by the dotted line in the figures is limited by climb performance. The increase in fuel consumption and the reduction in the number of passengers at the lower altitudes and higher cruise speeds are adverse factors affecting operational costs.

The data for the economic trade study is summarized on Table XXVII. The block fuel used in the DOC analysis is based on 55% payload factor, which is considered representative of airline operations. The airframe price, which includes nacelles, is the aircraft flyaway price less the price of engines.

Increase	DOC Change %
10% Fuel Price	2.4
10% Airframe Price	4.1
10% Engine Price	1.9

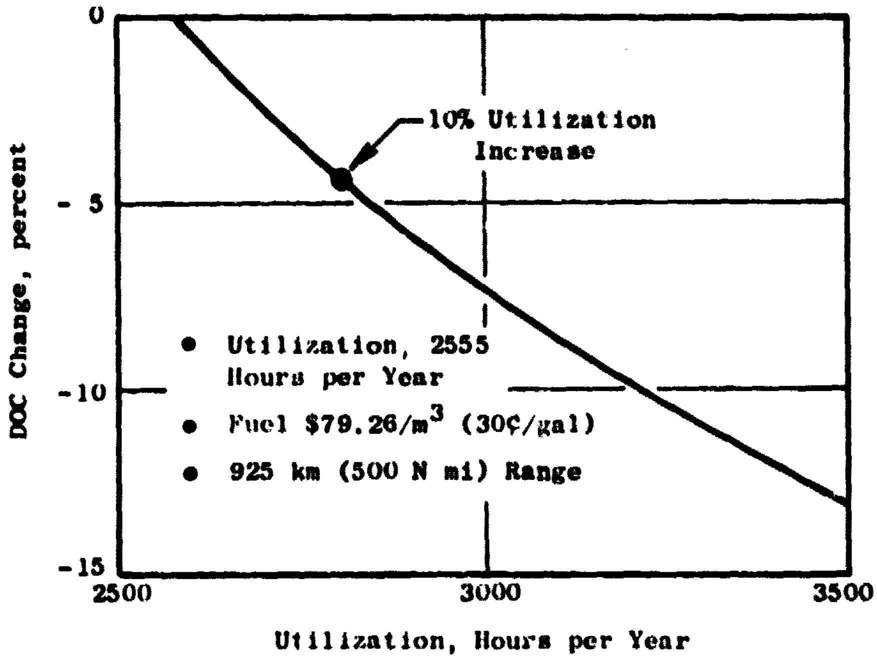

Figure 68. Operating Cost Sensitivities.

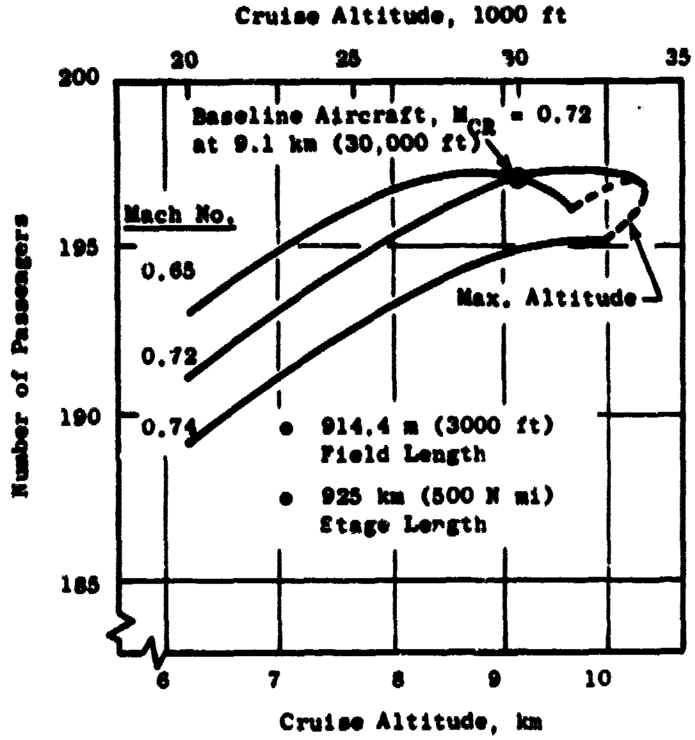

Figure 69. Payload Capabilities.

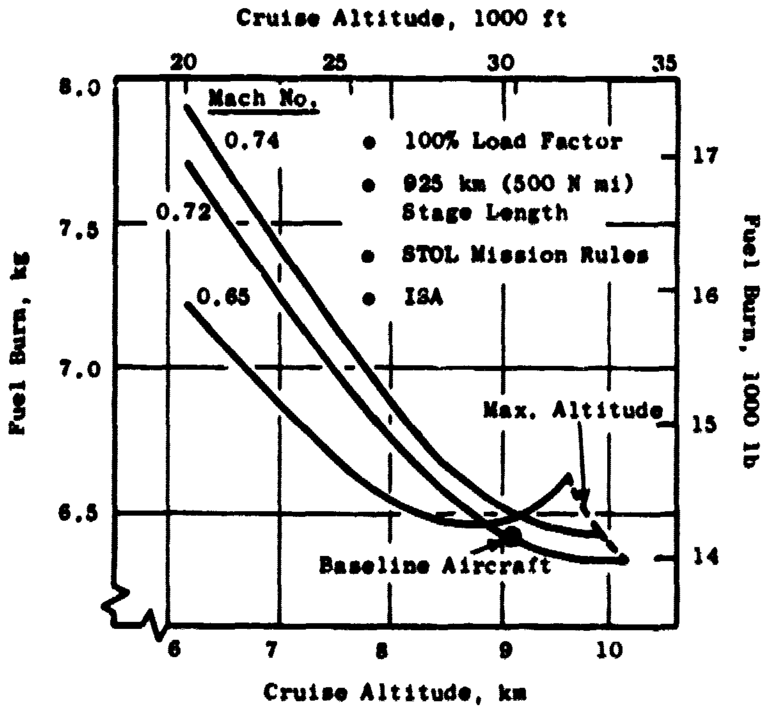

Figure 70. Block Fuel.

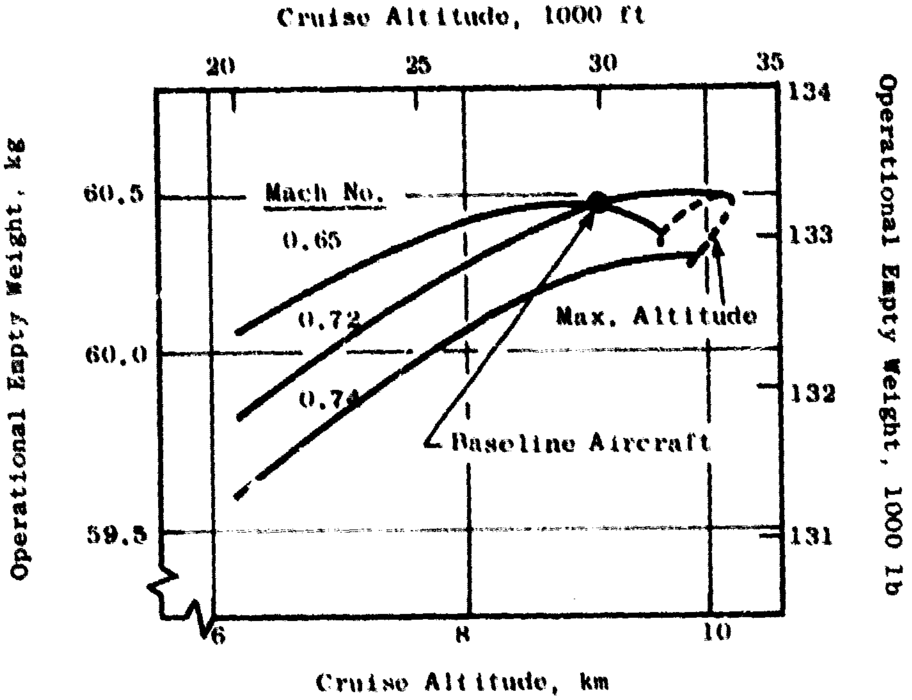

Figure 71. Operational Empty Weight.

Table XXVII. Data for Economic Trade Analysis

Block Fuel: see Figure 72

Block Time: see Figure 73

Boeing 1975 STOL Economic Rules and Dollars

Utilization: 2555 Block Hr/Yr

2-Man Crew

Fuel: \$79.26/m^3 (30¢ per U.S. Gallon)

Study Prices: (1975 Dollars)

- Airframe: see Figure 74 (includes \$403,000 per nacelle)
- Engine: \$1,140,000 each
- Spares: 6% Airframe 30% Engines

Depreciation: 14 Years to 2% Residual Value

Insurance: 1% of Flyaway Price

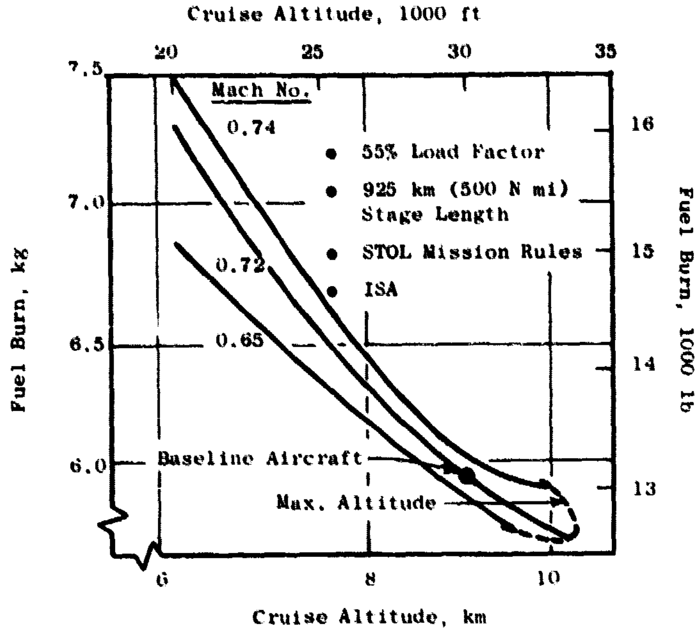

Figure 72. Block Fuel.

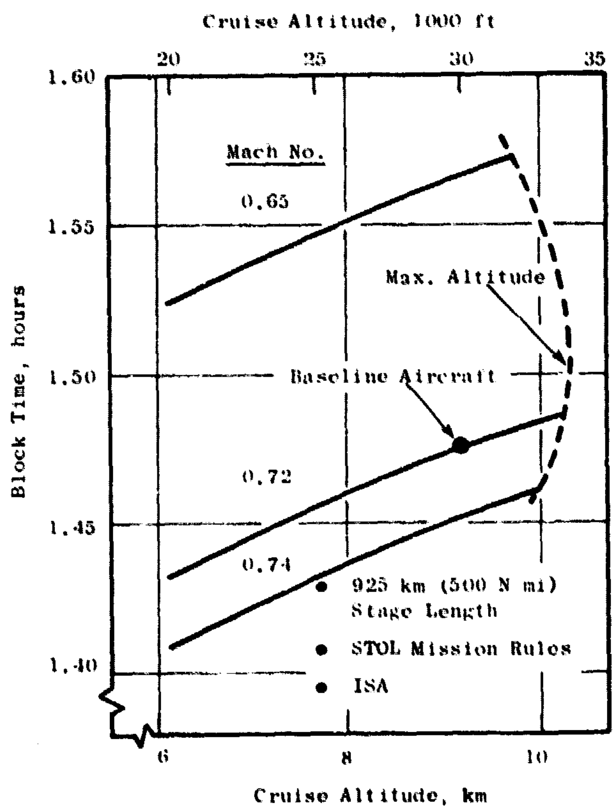

Figure 73. Block Time.

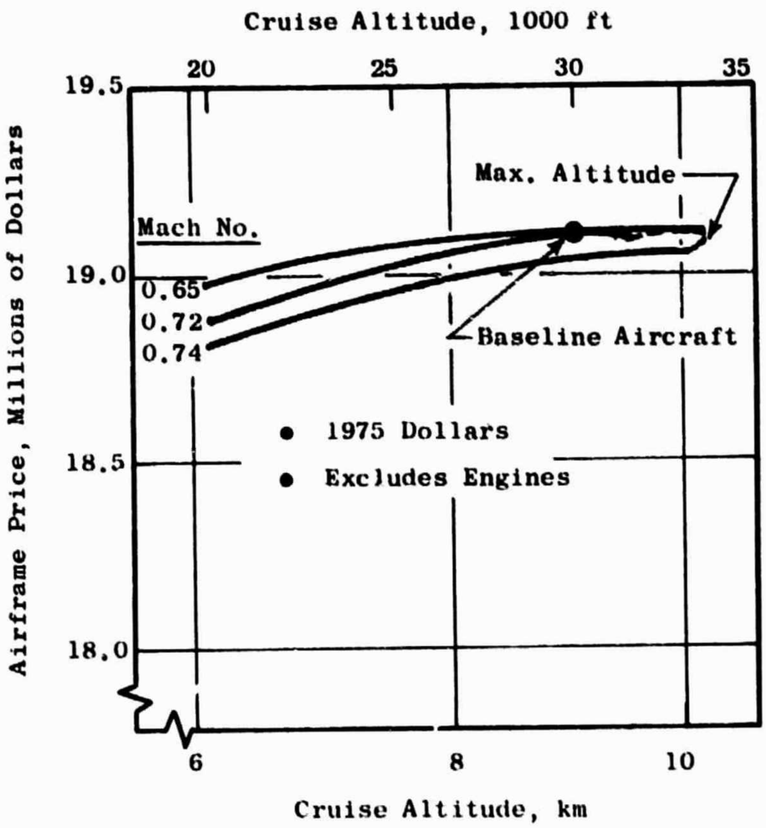

Figure 74. Airframe Price.

The results of varying the cruise altitude and Mach number on the direct operating cost is shown on Figure 75. The DOC is expressed in cents per available seat statute mile. The relative effects on the baseline aircraft are shown in Figure 76. The rise in the DOC at the lower altitudes is caused primarily by the increased fuel consumption due to cruising at nonoptimum aircraft ML/D. A secondary effect is the reduction in the number of passengers (available seats) as a result of the increased fuel consumption.

The effect of varying cruise Mach number at constant altitude can be seen on Figure 77. The cost elements which comprise the DOC for the baseline airplane are shown by the center bar. If the cruise speed is raised the cost of the increased fuel consumption is slightly greater than the cost benefits acrued from depreciation, maintenance, and insurance, which are amortized over a larger number of yearly flights. If the cruise speed is lowered, the benefit in fuel cost is insufficient to offset the increased cost of the other elements which are amortized over a smaller number of yearly flights.

The DOC and fuel consumption optimize at a Mach number of 0.72 and a maximum cruise altitude of 10.3 km (33,700 ft). This compares to a cruise altitude of 9.1 km (30,000 ft) and a cruise Mach number of 0.72 which were used for the design conditions of the baseline airplane.

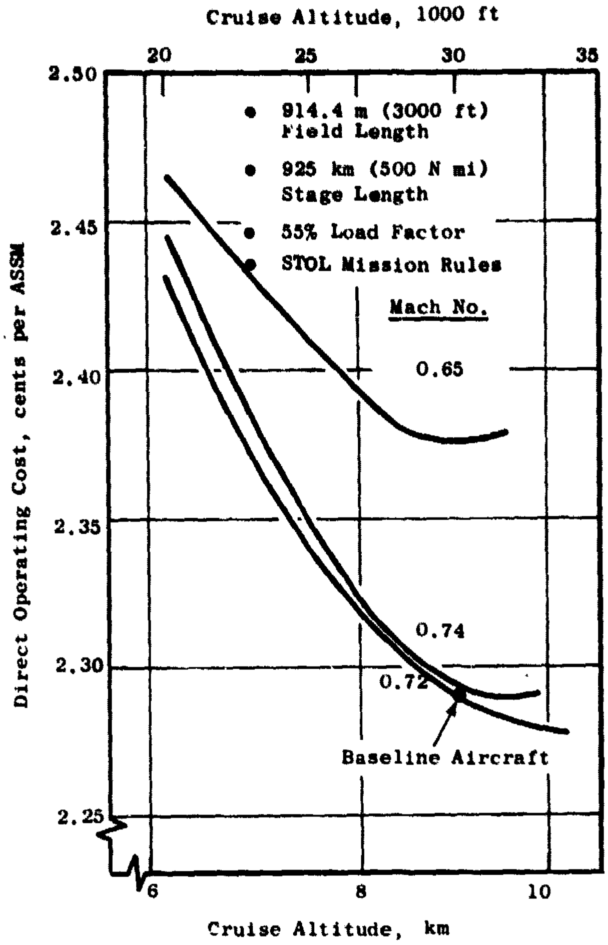

Figure 75. Direct Operating Cost.

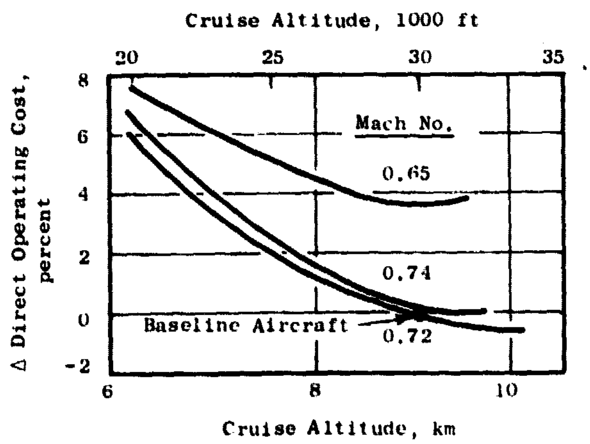

Figure 76. DOC Change from Baseline.

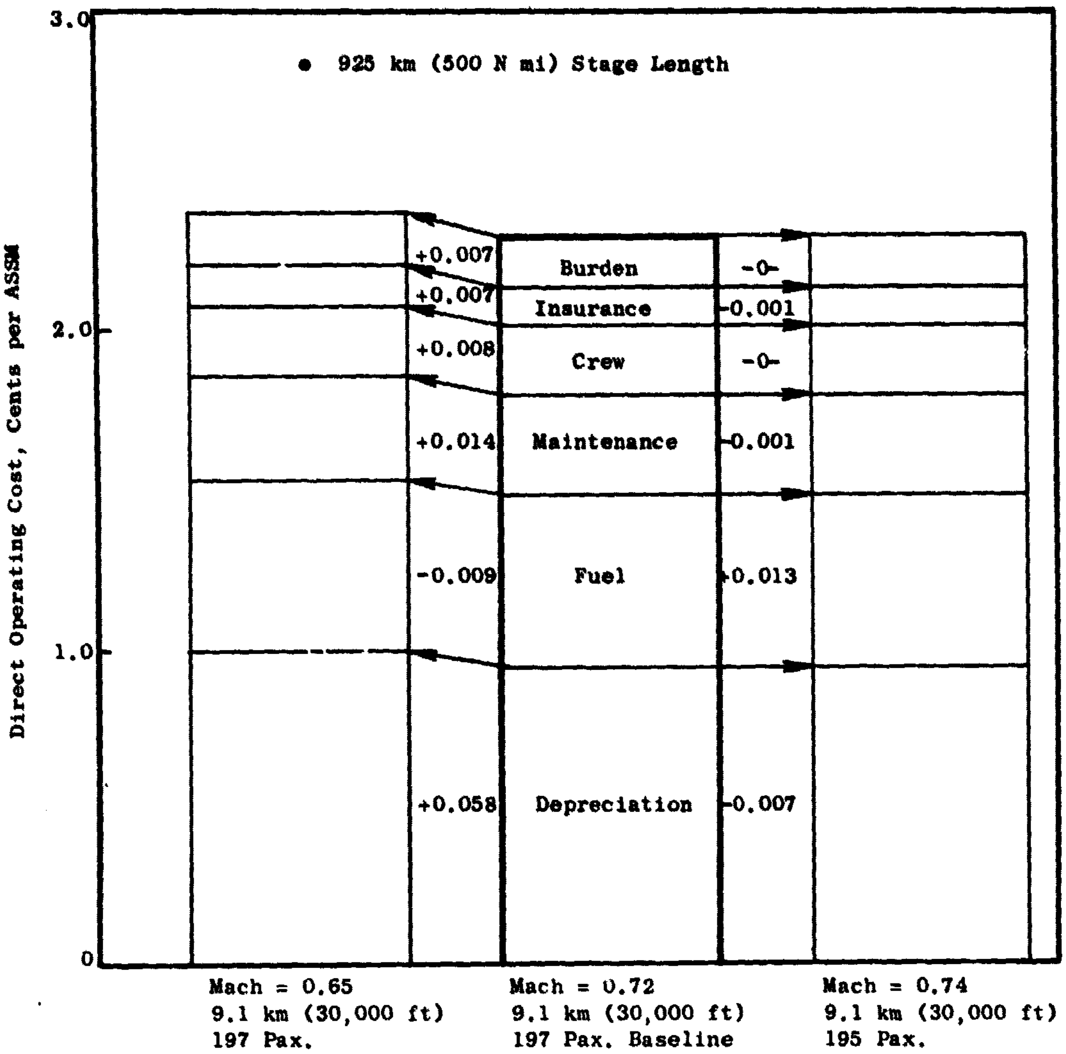

Figure 77. Direct Operating Cost Elements.

9.0 CONCLUSIONS

The system integration studies have strengthened the conclusions reached in earlier conceptual studies that a production derivative of the OTW QCSEE system promises to be an economical propulsion system for short-haul, STOL aircraft using powered lift. The increased concern about energy conservation, compounded by escalating fuel prices, has decreased the emphasis on increasing cruise speeds. Studies are being addressed to advanced technology CTOL aircraft that achieve significant reductions in fuel usage at current and reduced cruise speeds.

Specific conclusions applicable to a short-haul airliner fitted with engines based on utilization of the OTW/QCSEE technology, that can be drawn from this study are these:

- A "D"-shaped exhaust nozzle incorporating area control doors can be designed to meet the exhaust area requirements and, also, to provide sufficient exhaust spreading over the wing to achieve desired powered lift performance.

- The upper exhaust nozzle wall can be used to provide a thrust reverser blocker, which, with suitable lip and side skirt geometry, will provide the desired 35% thrust in reverse.

- Electronic controls offer the means by which the variable-geometry features and different modes of operation can be accommodated without increased pilot work load.

- The thrust available for takeoff, climb, and cruise is well matched to the requirements of the baseline short-haul, powered-lift airliner and consistent with the projected commercial aircraft requirements of the mid 1980's.

- The inlet geometry required for satisfactory inlet operation dictates an internal lip radius of $R_{HL}/R_i = 1.21$ and a lip shape in form of 2:1 ellipse. An external cowl having $R_{HL}/T_{max} = 0.90$ and cowl length ratio of $X/D_{max} = 0.22$ is required.

- This external cowl shape will provide good internal performance at high angle-of-attack conditions and also exhibit low external drag at cruise and at engine-out, takeoff-climb conditions. This can be attained with a cowl having a maximum diameter no larger than that of the QCSEE integral fan frame.

- Stopping the baseline airplane on a 914.4 m (3000 ft) long runway without brakes requires a thrust reversing effectiveness ≥ 0.35.

- Structural acoustic panels and the use of composite materials results in significantly reduced nacelle component weights - QCSEE-type inlet 25 to 20% lighter than equivalent metal "state-of-the-art" inlet.

- The thin-wall nacelle and relatively small maximum nacelle diameter for a 180.3 cm (71 in.) diameter fan result from the high Mach inlet and the integration of nacelle and engine components.

- A radiation shield, plus under cowl fan cooling air, is required in the rear area of the core cowl to keep the cowl door temperatures within the operating limits of polymeric composites.

- Frequent exposure of nacelle components to the possibility of incidental damage resulted in the selection of Kevlar 49 because of its greater toughness. Glass and/or graphite is used in areas requiring higher strength or stiffness.

Special areas of interest requiring further development and/or demonstrations:

- Wind tunnel testing is required to optimize the exhaust nozzle/thrust reverser geometry. Requirements for low cruise drag, sufficient exhaust turning over the flaps in the powered lift mode, and adequate thrust effectiveness in the reverse mode are highly dependent upon the detail aircraft design. Proper integration of the propulsion system and the aircraft will require further effort, including consideration of vortex generators on the wing surface, side skirts on the thrust reverser blocker, and increased negative cutback on the area control doors.

- Characteristic of thrust transient during change from approach to reverse thrust requires exploration with a moveable thrust reverser.

- Further study of geared engine heat rejection rates and fuel tank heat dissipation capacity is needed to ensure adequate oil cooling at all flight conditions.

10.0 REFERENCES

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APPENDIX A

AMERICAN AIRLINES
OPERATIONAL SCENARIO AND GENERAL REQUIREMENTS
FOR MULTIENGINE STOL PASSENGER TRANSPORT
AIRPLANE FOR INTRODUCTION IN 1980-1982

This document was used as a guideline in the baseline aircraft sizing study.

February 13, 1974
(Revised 4/10/75)

1.0 GENERAL

The following is intended as a guide to airplane/engine manufacturers submitting design studies for a large multiengine, short-medium range, high reliability transport. These requirements will be used as a basis for evaluating the qualities of such designs and each submittal should address itself to them. In the event a particular design does not meet one or more of these requirements, cogent reasons and justification for noncompliance should be included in the submittal.

The basic configuration of the airplane shall be tailored for the carriage of passengers. The economy and utility of the airplane should not significantly be compromised by considerations for carriage of cargo.

With due regard to the restraints imposed by the detailed recommendations made below, the airplane is expected to have a capacity of 150 to 170 passengers in a 100% Coach Class at 91.4 cm (36 in.) pitch. The airplane will be used to expand commercial airline service into smaller airports where such service is not currently available and it must therefore be as quiet as is technically reasonable.

2.0 MISSION REQUIREMENTS

The airplane should be capable of operating regularly non-stop design missions, similar to the following, with a full load of passengers, plus their baggage and 1587.6 kg (3500 lb) of cargo in belly containers.

(a) New York - Detroit

(b) New York - Chicago

The requirements given below give quantitative technical definitions consistent with the above policy.

2.1 The airplane should be capable of carrying a payload of (N x 200) + 1587.6 kg (3500 lb) - where N is the number of passenger seats in a full coach class interior - over the ranges and under the conditions tabulated on the following page.

2.2 Fuel Reserve (Standard)

The total fuel remaining over intended destination shall be assumed to be the sum of the following fuel quantities:

1. Fifteen minutes holding at 1.52 km (5000 ft), at best holding speed, calculated at the predicted destination landing weight.

2. Commencing at predicted destination landing weight, climb from sea level at the preferred climbing speed and cruise at not less than 4.57 km (15,000 ft) at preferred Mach number for a total diversion distance (climb and cruise) of 277.8 km (150 N Mi). No descent allowance.

	A NYC-DTW	B NYC-CHI
Equivalent Still Air Distance (NM)	1250 km (675 N Mi)	1250 kg (675 N Mi)
Assumed T/O Runway Length Available	914.4 m (3000 ft)	Greater than 914.4 m (3000 ft)
Takeoff Elevation	0	0
Takeoff Ambient Temperature	305.6 K (90° F)	305.6 K (90° F)
Enroute Ambient Temperature	ISA + 10° C	ISA + 10° C
Cruise Mach Number	0.65 Min	0.65 Min
Cruise Atitude	Not Less Than 7.62 km (25,000 ft)	Not Greater Than 9.45 km (31,000 ft)
Full Reserves	See Para 2.2	See Para 2.2

*Note: Cruise Mach number should be optimized on the basis of maximum seat miles per 378.5 m^3 (100,000) gallon of fuel consumed.

3. Forty-five minutes holding at 1.52 km (5000 ft), at best holding speed, commencing at the weight corresponding to the end of diversion cruise.

2.3 Fuel Capacity

Sufficient fuel tankage at 779 kg/m^3 (6.5 ppg) shall be provided in the wing structure to permit loading the airplane to maximum takeoff weight at a zero fuel weight equivalent to a weight no greater than the EOW plus 50% of the space-limited payload.

2.4 Fuel Tank Location

No fuel tankage shall be permitted outside the wing box in the underfloor area of low wing designs or center wing areas of the aircraft on high wing designs.

3.0 EXTERNAL NOISE CHARACTERISTICS

3.1 Noise

The airplane shall be capable of complying with the appropriate noise regulations as called for by Federal statutes at the time of application for an original type certificate or when revenue service is initiated, whichever are the more stringent.

3.2 The takeoff climb flight paths used to show compliance with the above requirements shall not require a pitch angle, that is, the angle between the passenger cabin floor and the horizontal, greater than that necessary to achieve aircraft optimum climb performance (goal 20 degrees maximum).

3.3 The overall sound pressure levels and speed interference levels in the control cabin and passenger cabin, at altitudes and speeds specified below, and with representative engine cruise thrust on both engines typical airline interior configuration, and with standard materials should not exceed the following values:

Pressure Altitude, km, (ft)	9.4	(31,000)	6.1	(20,000)
Speed	0.74	Mach		0.65
db	OASPL	SIL	OASPL	SIL
Pilot's Seat, Head level	85	67	90	74
First Class Window Seats, Head Level	82	62	81	69
Tourist Class Window Seats, Head Level	90	65	89	69

4.0 APPROACH AND LANDING PERFORMANCE

4.1 The following requirements shall be met in the landing configuration with maximum landing flaps and at maximum landing weight with and without propulsive lift:

1. Stall speed ($V_{S_{1g}}$) shall not exceed (TBD) Kts. EAS.

5.0 POWER PLANTS

Engines of the high bypass fan type in the 88,960 N (20,000 lb) thrust size are recommended. Maintenance features directed at minimizing the engine servicing time are necessary. Special attention shall be given to the starting system, reverser system and instrumentation to maximize reliability. The cost for a delay shall be considered to be 300 dollars for up to 10 minutes, 700 dollars for 20 minutes, and 2000 dollars for 30 minutes or over. The requirements, set forth in NASA CR-12134, as amended for STOL peculiar requirements, shall be used as a general guide in system design and program planning. Final engine design and selection must give due consideration to the following:

a. Long-term specific fuel consumption performance retention capability
b. Noise characteristics
c. Pollution emission (visible and invisible) as specified by EPA and/or FAA requirements
d. Durability, reliability

e. Low maintenance costs
f. Applicability to other aircraft as a total propulsion system
g. Long-term warranties

5.1 The engines should be able to provide rated takeoff thrust automatically at full throttle, at any ambient temperature and altitude, approved for takeoffs. The aircraft shall also be certificated for reduced thrust takeoffs of not less than 10% of rated thrust with a goal of 20% rated thrust according to the available performance margins. Certification and appropriate power setting procedures for reduced power takeoffs shall cover ambient temperatures up to at least ISA + 5° C (41° F), and thrust levels from maximum takeoff thrust down to maximum climb thrust for all takeoff altitudes. It shall be possible to obtain the reduced takeoff thrust with the throttle at the takeoff thrust within two seconds from the appropriate ground and flight idle positions if desired.

5.2 It shall be a design goal to actuate the thrust reversing system from the forward full-thrust position to the reverse full-thrust position, and vice versa, within one second at all speeds on the ground, up to the maximum touch down speed for a no-flap landing, and to obtain maximum available thrust relative to the thrust lever position within a total elapsed time of five seconds. In addition, actuation out of reverse in flight shall be possible up to at least 102.9 m/sec (200 knots) indicated airspeed at all altitudes. Due consideration must be given to the impact of reversible pitch fans and the consequences of inadvertent reversal in flight and suitable protection measures must be provided. Consideration should also be given to the use of reverse thrust in flight for achieving aircraft deceleration and high rates of descent, particularly in the approach mode.

5.3 It is particularly important that the engines be protected against ingestion of foreign objects during all phases of flight operations. Resorting to operating procedures to provide this protection, particularly during thrust reverser operation, shall be minimized.

6.0 OVERALL DIMENSIONS

The overall dimensions of the airplane should be in the order of:

- a wing span of 32.9 m (108 ft)
- an overall length of 48.8 m (160 ft)
- an overall height of 17.5 m (57-1/2 ft)

7.0 GROSS WEIGHT/AIRPORT COMPATIBILITY

At a weight 5% in excess of the ramp weight required to perform mission 'B', with the center of gravity in the most adverse location, and with all other relevant items in the most adverse configuration (e.g., tire pressure, etc.), the airplane shall be capable of operating on all taxiways,

runways, (including pier structures), aprons, etc., at LGA airport, without restrictions in maximum allowable taxi or takeoff weight due solely to pavement loading limitations.

7.1 All external servicing connections shall be physically interchangeable and compatibly located on the airplane with respect to fixed and portable airport facilities which will service B727 and DC-9 aircraft.

7.2 All passenger, cargo, and galley service doors shall be positioned horizontally and vertically such that they are compatible with fixed B727 and DC-9 terminal facilities.

8.0 INTERIOR ARRANGEMENT

8.1 Provisions shall be made to vary the percentage mix between First Class and Coach passengers and to vary the seat pitch. First Class/Coach ratios between 0 and 30% should be considered, with seat pitches varying between 81.3 cm (32 in.) and 101.6 cm (40 in.).

8.2 A single deck passenger cabin is preferred.

8.3 Passenger compartment aisle(s) should not be less than 50.8 cm (20 in.) wide.

8.4 Seat units may incorporate more than 2 seats; i.e., the basic interior may include triple seats or adjacent double and/or single seats. If triple seats are used, minimum overall seat width should be not less than 165.1 cm (65 in.) [i.e., 747 triple seat]. Consideration should also be given to increased seat pitch, and/or improved seat design, to provide equivalent passenger comfort and adequate in/out access for window seat passengers. Adjacent seat units, if installed, shall not be closer than (TBD) inches, between armrests, to provide for tables and stowage.

8.5 Emergency evacuation provisions should meet all requirements of the latest FAR.

8.6 An Adequate number of lavatories should be provided. The formula N-20/40 + 1, where N is the total number of passenger seats in the basic mixed class configuration, provides guidance on the number required.

8.7 Galley capacity, where required, should be sufficient to provide meal service to the same standards as provided on B727 and DC-9 on a per passenger basis. The design should consider handling all food service from carts.

8.8 Coat hanging space equivalent to 1.8N cm (0.7N inches) should be provided, where N is the total number of passenger seats in the basic mixed class configuration.

8.9 Consideration shall be given to the incorporation of under-seat, overhead and special interior baggage storage racks for all baggage on the short-haul service. These provisions should be arranged to permit rapid and convenient access by enplaning and deplaning passengers using either one pair or two pairs of opposing passenger entry doors.

8.10 An airstair system shall be provided to rapidly load and unload passengers.

9.0 BAGGAGE AND CARGO HANDLING

All passenger baggage shall be considered to be of the carry-on type. A preloaded cargo handling system shall be provided to accommodate 14.15 m^3 (500 ft^3) of cargo. Consideration should be given to having the system capable of accepting LD-3 type containers. If the containers specified for the airplane are not actually LD-3 type containers, they should be interchangeable with the LD-3 type, and must be capable of being carried by DC-10, B747 and L-1011 aircraft.

9.1 The cargo containers shall be fuselage structure constrained without the requirement for locking devices, except for a doorway roll-out-stop.

9.2 The cargo handling/container system shall be designed so that containers can be loaded and unloaded by one man at each compartment opening, external to the opening.

9.3 Cargo doors shall be outward opening, canopy type, on the right-hand side of the airplane and incorporate positive latching to prevent opening in flight.

10.0 OPERATION FEATURES

10.1 The aircraft may normally operate from a parallel parked position with ground level passenger loading. Consideration should also be given to making the aircraft compatible with existing nose-in parking, second level loading for B727 and DC-9 aircraft.

10.2 The airplane shall be capable of a 20 minute through flight, or a 30 minute turn as a maximum.

10.3 Consideration shall be given to a self-contained means for backing the aircraft from its parked position.

10.4 Ground Maneuvering

1. The aircraft shall be capable of easily executing a 180° turn within a maximum pavement width of (TBD) feet, at its maximum taxi weight, with its center of gravity in the most adverse

location. The radius of the circle described by the extremity of the aircraft's planform under these conditions shall not exceed (TBD) feet. Best efforts shall be exercised to achieve significantly lower turning radii than these specified maxima. Additionally, it shall be a design goal to achieve these characteristics with a nose landing gear position located as nearly as possible below the pilots' seats to aid in visually taxiing the aircraft.

2. The aircraft shall be capable of executing the turn of paragraph 10.4.1 from a standing start on a level dry pavement, with cold tires, at thrust levels that will not cause jet blast velocities in excess of 35.0 m/sec (115 fps) along a line perpendicular to the aircraft centerline passing through the rearmost point on the aircraft, or 25.9 m/sec (85 fps) along a line perpendicular to the aircraft centerline, and 15.2 m (50 ft) aft of the rearmost point on the aircraft. It shall be a design goal to achieve levels no greater than 27.4 m/sec (90 fps) and 18.3 m/sec (60 fps) at these locations, respectively.

10.5 Special consideration shall be given to providing redundancy and fault isolation capability in the aircraft systems to an extent sufficient to ensure that mechanical delays are minimized, and to provide operational characteristics, performance capability, and airworthiness characteristics with all engines operating comparable, insofar as practical, to a four-engine aircraft with all engines operating, and in the case of an engine out, to a four-engine airplane with one engine out.

10.6 Any major component (i.e. engine, LRU, etc.) should be replaceable within a four-hour period.

10.7 The airplane shall be designed for a V_{mo}/M_{mo} of not less than (TBD).

10.8 The airplane shall be designed to be fully operational by a two-man cockpit crew, but with separate and sufficient controls and systems monitoring devices for adequate functioning of a third crew member. At least one observer seat should also be provided and should be suitably positioned for check-training supervision.

10.9 The airplane shall be equipped with automatic spoiler extension for landing and aborted takeoffs, and with automatic safety retraction for go-around. The system should be similar to that of the B747-123.

10.10 Cockpit design shall pay particular attention to the requirements of SAE ARP 268D, "Location and Actuation of Flight Deck Controls for Commercial Transport Type Aircraft"; and SAE AS 580A, "Pilot Visibility from the Flight Deck - Design Objectives for Commercial Transport Aircraft."

The airplane shall be designed to display the highest state of the art in flying qualities and be certified for Category II operation prior to initial delivery. In addition, the system should be designed to anticipate Category III operations by being configured to facilitate postdelivery modification, or incorporation of the following features:

- Best state-of-the-art autoland autopilot system (preferably fail operative)

- Control (force) wheel steering

- Automatic directional and lateral control guidance throughout takeoff and landing roll

- Automatic pilot go-round, pilot initiated

- Provisions for dual, windshield glass type flight director heads-up display

- Automatic wheel braking system, pilot adjustable for desired deceleration

- Electronic attitude director indicator incorporating visibility enhancement presentation. Color CRT preferred for symbology.

10.11 The main landing gear shall incorporate features designed to make the airplane land soft. It should be possible to land the airplane at substantially higher sink rates than are average on current jet aircraft without incurring greater than average normal vertical accelerations (G's). A minimum normal acceleration of 0.4G at a minimum sink rate of 1.5 m/sec (5 ft/sec) is suggested as a criterion.

11.0 OTHER REQUIREMENTS

11.1 The maximum Design Zero Fuel Weight shall not be less than the sum of 110% the Operating Weight Empty and the space limited payload. The space limited payload is defined as: 200N + 10 [total containerized + bulk (if any) baggage and cargo volume] where N is defined as the total number of passenger seats in the basic mixed class configuration and baggage/cargo volume is in cubic feet.

The maximum Design Landing Weight shall not be less than the sum of 110% of EOW plus space limited payload plus the fuel reserves of paragraph 2.2. Consideration should also be given to the possible requirement to through fuel at one or more stations for operational or economic reasons.

11.2 Cabin differential pressure shall not be less than (TBD) psi.

11.3 For performance calculation purposes, the total cabin air bleed in cubic feet per minute will be assumed to be 20 times the number of passenger seats in the maximum coach configuration. Power extraction corresponding to an electrical load of 100 kW should be assumed.

11.4 Space and wiring provisions should be included for a pictorial map-type display centrally located and visible to both pilots, and which could be operated by either of two area-navigation type computers.

11.5 Space, wiring, and structural provisions should be included for a performance recorder and maintenance monitoring system.

11.6 High intensity coded anticollision condenser discharge-type external lighting shall be provided.

11.7 Consideration should be given to assuring the quietness of operation of all systems and mechanically functioning components of the airplane as perceived within the airplane.

11.8 The aircraft should be equipped with inflight speed brakes which will not induce longitudinal or lateral trim changes, or otherwise adversely affect flight characteristics, over the entire design flight envelope. Such brakes shall be usable at all airspeeds and will not result in noticeable or objectional buffeting.

11.9 Consideration should be given to incorporating an inerting system to afford fire and explosion protection to the under-floor cargo compartments, wheel wells, fuel tanks, vents, and engines. Consideration should also be given to the use of nitrogen for maintaining oxygen in a liquid state, should cryogenic systems be incorporated.

11.10 All elements of the engine control system and aircraft flight control system which either pass through or are adjacent to the cabin floor or ceiling shall be protected against the catastrophic consequences that may arise from failure or significant deformation of the floor or ceiling.

12.0 LOADABILITY

Using the passenger seating and cargo/baggage loading assumptions defined below and with operating items in normal location, subject to limitations on total load set by design weights, the aircraft center of gravity shall remain within certified limits for takeoff, flight, and landing under the following conditions:

a. Any number of passengers from zero to maximum in the First Class compartment, plus

b. Any number of passengers from zero to maximum in Coach compartment, plus

c. 13.6 kg (30 lb) of carry-on baggage per passenger for any passenger load, plus

d. Any quantity of cargo up to 1587.6 kg (3500 lb) to be distributed about the centroids of the forward and/or aft containers as required to maintain the center of gravity within limits, plus

e. Any quantity of fuel from zero to the maximum tank capacity, except that for takeoff, a fuel quantity of less than 4536 kg (10,000 lb) need not be assumed.

The following passenger seating arrangements shall be covered, with window seats occupied first, aisle seats next, and remaining seats last.

1. Passengers in each compartment, loaded from front to rear.

2. Passengers in each compartment, loaded from rear to front.

FAR regulatory allowances shall be made for adverse passenger and crew movement in flight and for gear and flap retraction and extension, as appropriate. Passengers are assumed to weigh 77.1 kg (170 lb) each. Baggage and cargo stowage density is assumed throughout to be 160.3 kg/m^3 (10 lb/ft^3). The forward and aft containerized baggage compartments are assumed to house a full complement of baggage containers.

12.1 Aircraft tip-up characteristics, both rolling and static, will be such that no special precautions in loading, unloading, or in operating procedures, or in the use of special ramp equipment will be required during normal airline operation.

12.2 An on-board weight and balance system shall be incorporated that will provide an instantaneous read-out of gross-weight and center of gravity position. Full-scale accuracies of approximately 0.08% are suggested as a design goal. The system shall also provide instantaneous visual indication of landing impact load, including a sustained (resetable) maximum impact load. Consideration should be given to utilizing the flight/maintenance recorder for this purpose.